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RESEARCH MEMORANDUM

ALTITUDE WIND TUNNEL INVESTIGATION OF THE PROTOTYPE J40-WE-8 TURBOJET ENGINE WITHOUT AFTERBURNER

By John E. McAulay and Harold R. Kaufman

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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SUMMARY

An investigation was conducted in the Lewis altitude wind tunnel to evaluate the performance characteristics of the prototype J40-WE-8 turbojet engine without an afterburner. Data were obtained with an electronic control operative and inoperative. The performance data were obtained at altitudes from 15,000 to 60,000 feet and flight Mach numbers of 0.17 to 1.68.

Fixed-exhaust-nozzle data showed that in general increasing altitude resulted in an increase in corrected net thrust at a given corrected engine speed. These data also showed that above a corrected engine speed of 7000 rpm a change in altitude at a given corrected engine speed had no effect on the corrected air flow. A method is presented to define the effect of changes in engine operating and flight conditions on the pumping and air-flow characteristics and the combustion efficiency. This made it possible to calculate thrust and fuel flow for conditions other than those at which the data were obtained. These calculated values were in close agreement with values obtained in the direct investigation.

INTRODUCTION

As part of a comprehensive investigation of the J40 turbojet engine conducted at the NACA Lewis altitude wind tunnel, the steady-state engine performance of the prototype J40-WE-8 turbojet engine without afterburner was obtained and is presented herein. Preliminary performance tests of an earlier model, the XJ40-WE-6, revealed a severe surge condition in the compressor at high corrected engine speeds (reference 1). A basic redesign of the compressor and other modifications in the compressor and the combustor were incorporated in the XJ40-WE-6 turbojet engine (references 2 and 3). In this report the modified engine is designated "the prototype J40-WE-8 without afterburner."

Performance data presented herein were obtained over a range of engine speeds at five fixed settings of the variable-area exhaust nozzle. These data were obtained at altitudes from 15,000 to 45,000 feet and at

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flight Mach numbers of 0.62 and 0.99. Data were also obtained with an open exhaust nozzle at altitudes of 50,000 and 55,000 feet at a flight Mach number of 0.62. In addition, some data were obtained at flight Mach numbers as high as 1.68 at altitudes of 55,000 and 60,000 feet by a different method of simulation wherein engine-inlet temperature and pressure, but not tunnel static or altitude ambient pressure, are reproduced. The use of the engine pumping characteristics made it possible to calculate engine performance for a greater range of flight Mach numbers and altitudes than were experimentally investigated. 2733

The data obtained at fixed settings of the variable-area exhaust nozzle are presented in both graphical and tabular form. In addition, data with an electronic engine control operative are also presented in tabular form.

APPARATUS AND INSTALLATION

The prototype J40-WE-8 turbojet engine without afterburner has a static sea-level thrust rating of 7500 pounds at an engine speed of 7260 rpm and a turbine-inlet temperature of 1885° R (1425° F). At this operating condition the air flow is approximately 142 pounds per second. The engine components included a divided inlet duct (fig. 1), an eleven-stage axial-flow compressor, an annular combustor, a two-stage turbine, a tail pipe, and a variable-area exhaust nozzle. Without the afterburner the engine length is 186 inches and the maximum diameter 43 inches. The dry weight of the engine and accessories is about 3000 pounds.

The engine was mounted on a wing section that spanned the 20-foot-diameter test section of the altitude wind tunnel (fig. 2). Dry refrigerated air was supplied to the engine from the tunnel make-up air system through a duct which was divided and connected to the engine inlets. Throttle valves installed in the main duct permitted regulation of the pressure at the engine inlet.

Engine thrust and drag measurements by the tunnel balance scales were made possible by the frictionless slip joint located in the main duct upstream of the engine. Instrumentation for measuring pressures and temperatures was installed at various stations in the engine (fig. 3). Pressure measurements at the exhaust-nozzle inlet were available for only a small portion of the investigation. Turbine-inlet radial temperature distributions were determined by ten traversable sonic-flow thermocouple probes.

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PROCEDURE

Engine performance data presented in this report were obtained at the flight conditions shown by the following table:

Altitude (ft)	Flight Mach number						
	0.17	0.62	0.92	0.99	1.19	1.46	1.68
15×10^3		x *					
35	x	x *		x *			
45		x *					
50		x					
55		x ✓	✓		✓		✓
60		✓	✓		✓	✓	✓

x control data

* fixed exhaust-nozzle data

✓ rated speed, "military" and "normal" turbine-inlet temperatures

The control scheduled data included open-exhaust-nozzle operating lines. The fixed-exhaust-nozzle data were obtained at projected exhaust-nozzle areas of 367, 421, 449, 479, and 535 square inches at several engine speeds for each exhaust-nozzle area. The fixed-exhaust-nozzle data are given in table I. Similarly, the control data are given in table II but are not presented graphically because standard inlet temperatures could not be maintained for several flight conditions.

In order to obtain the various flight conditions, the air flow through the make-up air duct was throttled from approximately sea-level pressure to a total pressure at the engine inlet corresponding to the desired flight Mach number at a given altitude. For most of the runs, the tunnel pressure was set at the desired altitude ambient pressure. In the calculation of flight Mach number, complete ram-pressure recovery at the engine inlet was assumed. The temperature of the inlet air approximated NACA standard values except that the minimum temperature obtained was about 440° R. The engine fuel used was MIL-F-5624 having a lower heating value of 18,700 Btu per pound and a hydrogen-carbon ratio of 0.171. The fuel temperature entering the engine fuel system was about 80° F.

The altitude at which standard altitude pressure could be maintained is limited by exhauster capacity. To extend the range of the

investigation to higher flight Mach numbers and altitudes, a technique was used wherein the engine performance could be obtained irrespective of tunnel pressure, as long as the tunnel pressure was less than the exhaust-gas total pressure. The engine-inlet pressures and temperatures which would exist at these flight conditions were reproduced while the pressure altitude in the tunnel test section was maintained at any convenient value. The variable-area exhaust nozzle was adjusted as necessary to obtain the desired values of engine temperature ratio. As indicated in reference 4, for given engine-inlet conditions and fixed 2733
engine speed, the engine air flow, fuel flow, and pressure ratio are not dependent on the ambient-air pressure for operation at a given engine-temperature ratio. The thrust was calculated from measured values of turbine-outlet pressure and temperature and engine air flow by the method given in appendix A.

RESULTS AND DISCUSSION

Generalized Performance

Typical engine performance data obtained at a flight Mach number of 0.62 and at two exhaust-nozzle areas are shown for altitudes from 15,000 to 55,000 feet in figure 4. The two exhaust-nozzle areas chosen were the largest and smallest at which a full range of engine speeds was obtained. These data have been corrected by the factors δ and θ derived in reference 5 and defined in appendix A.

The effect of altitude on corrected air flow is presented in figures 4(a) and 4(b). At corrected engine speeds above 7000 rpm, the data generalized to a single curve; however, at corrected engine speeds below 7000 rpm, the corrected air flow decreased as altitude was increased at a given corrected engine speed.

The corrected fuel flow (figs. 4(c) and 4(d)), the corrected specific fuel consumption (figs. 4(e) and 4(f)), and the corrected exhaust-gas temperature (figs. 4(g) and 4(h)) increased as altitude was increased at a given corrected engine speed.

Decreases in compressor and turbine efficiencies resulting from the lower Reynolds numbers at the higher altitudes required an increase in corrected enthalpy rise per pound across the engine to maintain the same corrected engine speed. Higher compressor pressure ratios resulted from the higher corrected temperatures at the turbine inlet (reflected by turbine-outlet temperatures). At high corrected engine speeds, the corrected air flow did not vary appreciably with compressor pressure ratio and no shift in the compressor characteristic curves occurred with altitude; hence, the corrected air flow generalized. At lower corrected engine speeds (below 7000 rpm), the effect of higher compressor pressure ratio and the shift in the compressor characteristics resulted in lower corrected air flows for higher altitudes.

Examination of the data shows that corrected enthalpy rise across the engine increased with altitude as a result of the higher corrected temperature rise across the engine even at low speeds where the corrected air flow decreased. This corrected enthalpy rise required an increase in corrected fuel flow. However, as the combustion efficiency is adversely affected by both high altitudes and low engine speeds (reference 6), the effect of altitude on corrected fuel flow (and corrected specific fuel consumption) will be even greater than would be expected from consideration of changes in corrected exhaust-gas temperature and air flow, especially at low corrected engine speeds.

Except at low corrected engine speeds, the corrected net thrust increased as altitude was increased at a given corrected engine speed (figs. 4(i) and 4(j)). Even at low corrected engine speeds this trend was evident at altitudes above 50,000 feet. These trends in corrected net thrust, which are similar to those shown in reference 7, are due to changes in corrected air flow, exhaust-gas temperature, and turbine-inlet pressure which are affected by decreased component efficiencies with increased altitude. At lower corrected engine speeds where the change in corrected net thrust with altitude is less (in some cases nonexistent) the decrease in corrected air flow offsets the increase in corrected exhaust-gas temperature and pressure.

Performance Maps

The engine performance maps presented in figure 5 were cross-plotted from data shown in figure 4 and similar data for other exhaust-nozzle areas. A map was constructed for each of the four flight conditions at which data for a full range of exhaust-nozzle areas and engine speeds were obtained. The coordinates of these maps are exhaust-gas temperature and engine speed with lines of constant net thrust, specific fuel consumption, and projected exhaust-nozzle area superimposed. Also shown are lines that indicate the exhaust-gas temperature that gives limiting turbine-inlet bulk and local temperatures. The limiting local turbine-inlet temperature is reached when the temperature at any radial position at the turbine inlet equals the manufacturer's specified limit for that particular radial position (reference 3). Curves shown above this latter limit were extrapolated.

The minimum specific fuel consumption encountered at these four flight conditions was about 1.20 pounds per hour per pound thrust and occurred at an altitude of 35,000 feet and a flight Mach number of 0.62 (fig. 5(c)). At the other flight conditions investigated, the minimum specific fuel consumption was about 1.25 pounds per hour per pound thrust. At high engine speeds, closing the exhaust nozzle from an area of 421 to 367 square inches resulted, in general, in an increase in specific fuel consumption. This increase is associated with a reduction in compressor efficiency as the compressor pressure ratio is increased (reference 2).

As total pressure at the engine inlet was reduced, the exhaust-gas temperature at which limiting turbine-inlet local temperature occurred approached the exhaust-gas temperature at which limiting turbine-inlet bulk temperature was encountered (fig. 5). As stated in reference 3, this is caused by the closer matching of the turbine-inlet temperature profiles with the manufacturer's specified profile as the engine-inlet total pressure was decreased. If the actual and the recommended profile were identical, the exhaust-gas temperature would, of course, be the same for either turbine-inlet limit. Because of mismatching of these profiles at low altitudes, only about 95 percent of the maximum net thrust possible could be realized without exceeding the local turbine-inlet temperature limit (fig. 5(a)). 2733

In the region above 75 percent of maximum net thrust for any flight condition, no large difference in specific fuel consumption was obtained for any particular schedule of exhaust-nozzle area and engine speed. Therefore, the exhaust-nozzle schedule used is not critical insofar as steady-state performance is concerned. Acceleration and thrust modulation are therefore the determining factors in the manufacturer's selection of an exhaust-nozzle schedule. The steady-state exhaust-nozzle schedule that allows the exhaust nozzle to remain open until rated engine speed is reached appears to give the best transient performance because: (1) the maximum rate of acceleration is possible, and (2) large increases in thrust may be obtained almost instantaneously by closing the exhaust nozzle at any engine speed. For example, at an engine speed of 6500 rpm, an altitude of 15,000 feet, and a flight Mach number of 0.62, it is possible to obtain about 55 percent thrust modulation. Using the previous example as a qualitative, but not quantitative guide, by operating with the exhaust nozzle open at the reduced thrust levels required during a landing approach or cruise condition, a large and almost instantaneous thrust increase is available in case of a "wave-off" or similar maneuver.

Use of Pumping Characteristics and Combustion Efficiency to Calculate Engine Performance

It is desirable to be able to calculate engine performance at flight conditions other than those presented in this report. In order to do this from pumping characteristics, it is necessary to define the

effect of a change in engine operating and flight condition on several engine parameters. To meet this requirement, the effect of Reynolds number on engine pumping and air-flow characteristics must be determined. It is also necessary that the variation of combustion efficiency and effective velocity coefficient of the exhaust nozzle be defined in terms of engine parameters that are readily available. In the following paragraphs these relations will be discussed and the curves necessary to calculate engine performance will be presented. It is important to
2733 note that engine pressure ratio does not include inlet-duct losses. Performance including duct losses may be calculated if these losses are known.

Engine air flow and pressure ratio. - Engine air flow and pressure ratio are shown as functions of engine temperature ratio for constant corrected engine speeds at a Reynolds number index of 0.222 in figures 6(a) and 7(a), respectively. Correction factors which account for the effect of Reynolds number on the air-flow and pumping characteristics are presented in figures 6(b) and 7(b). The correction factor for corrected air flow is the ratio of corrected air flow at the Reynolds number index in question to the corrected air flow at a Reynolds number index of 0.222. Similarly, the correction factor for engine pressure ratio is the ratio of pressure ratio at the Reynolds number index in question to the pressure ratio at a Reynolds number index of 0.222. Selection of the reference Reynolds number index (0.222 in this case) was made in order to utilize the high corrected engine speeds and engine temperature ratios investigated at this Reynolds number index.

Combustion efficiency. - Combustion efficiency is presented as a function of a combustion parameter W_aT_6 in figure 8. The restrictions imposed by the derivation of this parameter, which is given in appendix B, are that the corrected engine speed be about 75 percent of rated speed or greater, and that the engine temperature rise be 700° F or more.

Fuel flow. - With the assumption of unity combustion efficiency, engine temperature rise is plotted as a function of fuel-air ratio with lines of constant engine-inlet air temperature in figure 9 (data from reference 8). The use of this figure in conjunction with figure 8 makes it possible to calculate an actual fuel-air ratio. All the variables required to obtain fuel flow and ideal thrusts (no tail-pipe or nozzle losses) have been presented in figures 6 through 9.

Effective velocity coefficient. - An effective velocity coefficient given in figure 10 is required to calculate actual values of thrust. An explanation of the parameters used on this figure is given in appendix A.

A sample problem demonstrating the use of figures 6 through 10 is given in appendix C.

Engine Performance Obtained from Pumping Characteristics and Direct Experimental Data

Net thrust and fuel flow for the military and normal engine operating conditions are presented as a function of true airspeed for seven altitudes in figures 11 to 13. The data presented in figure 11 were calculated by means of the pumping characteristics and supplementary curves (figs. 6 to 10). Data presented on figure 12 were obtained from experimental data, using the method described earlier which avoids the necessity of duplicating flight ambient pressure in the tunnel test section. Figure 13 presents both experimental and calculated data. The experimental data shown in figures 12 and 13 were obtained at flight Mach numbers as high as 1.68. For military and normal conditions, the engine speed is 7260 rpm and the exhaust-gas temperatures are 1580° and 1440° R, respectively. These temperatures correspond to turbine-inlet temperatures of 1885° and 1750° R.

These data show that at low flight speeds (fig. 11(a)) the net thrust decreased as flight speed was increased from 0 to about 275 knots. Above flight speeds of about 275 knots (figs. 11 to 13), the net thrust increased with flight speed at an increasing rate up to a flight speed of about 900 knots. Further increase in flight speed resulted in a decrease in the rate at which net thrust increased (figs. 11(d) to 13). This latter trend is associated with the relation of inlet-air temperature to flight speed and the effect of reduced corrected engine speed and engine temperature ratio on the engine pressure ratio. Fuel flow increased with flight speed over the entire range of flight speeds.

A comparison of experimental data and data calculated from pumping characteristics is possible at an altitude of 60,000 feet (fig. 13). For the curves showing military operation, the maximum discrepancy in both net thrust and fuel flow is about 2 percent at high flight speeds. The curves showing normal operation are not in as close agreement, the maximum difference being about 4 percent at high flight speeds.

SUMMARY OF RESULTS

Fixed-exhaust-nozzle performance data were obtained at altitudes as high as 55,000 feet and flight Mach numbers as high as 0.99. In general, increasing the altitude resulted in an increase in corrected net thrust at a given corrected engine speed. Above a corrected engine speed of 7000 rpm, changing altitude at a given corrected engine speed had no effect on corrected air flow. However, below a corrected engine speed of 7000 rpm, the corrected air flow decreased as altitude was increased at a given corrected engine speed. For the four flight conditions at which engine performance maps were obtained, the minimum specific fuel consumption was about 1.20 pounds per hour per pound of thrust and occurred at an altitude of 35,000 feet and a flight Mach number of 0.62. The effect of exhaust-nozzle area and engine speed on specific fuel consumption was small at thrust levels above 75 percent of maximum. The selection of a schedule of exhaust-nozzle area and engine speed is therefore primarily dependent on the consideration of the acceleration characteristics.

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A method is presented to define the effect that a change in engine operating and flight condition would have on engine-pumping and air-flow characteristics, and combustion efficiency. This permits the calculation of net thrust and fuel flow for conditions at which data points were not obtained. These calculated values agreed closely with the actual values obtained. Curves of thrust and fuel flow for both military and normal operating conditions are shown for altitudes from 15,000 to 60,000 feet and flight speeds of zero to 1100 knots.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio

APPENDIX A

SYMBOLS AND METHODS OF CALCULATION

Symbols

The following symbols are used in this report: 2733

A	cross-sectional area, sq ft
B	thrust scale reading, lb
C_v	effective velocity coefficient, ratio of scale jet thrust to rake jet thrust calculated at turbine outlet
D	external drag of installation, lb
F_j	jet thrust, lb
F_n	net thrust, lb
g	acceleration due to gravity, 32.2 ft/sec^2
K	constant
M	Mach number
N	engine speed, rpm
P	total pressure, lb/sq ft abs
p	static pressure, lb/sq ft abs
R	gas constant, 53.4 ft-lb/(lb)($^{\circ}$R)
T	total temperature, $^{\circ}$R
t	static temperature, $^{\circ}$R
V	velocity, ft/sec or knots
W_a	air flow, lb/sec
W_g	gas flow, lb/sec
W_f	fuel flow, lb/hr
γ	ratio of specific heats

δ ratio of engine-inlet absolute total pressure to absolute static pressure of NACA standard atmosphere at sea level

η_b combustion efficiency

ρ density, slugs/cu ft

2733 θ ratio of engine-inlet absolute total temperature to absolute static temperature of NACA standard atmosphere at sea level

φ ratio of absolute viscosity of air at the engine inlet to the absolute viscosity of NACA standard atmosphere at sea level

$\frac{\delta}{\varphi\sqrt{\theta}}$ Reynolds number index

Subscripts:

e equivalent

eff effective

i indicated

r rake

s scale

0 free stream

1 inlet duct

2 engine inlet

3 compressor inlet

4 compressor outlet or combustor inlet

5 combustor outlet or turbine inlet

6 turbine outlet

7 exhaust-nozzle inlet

Method of Calculations

Flight Mach number. - The flight Mach number, when complete ram-pressure recovery was assumed, was calculated from the expression

$$M_0 = \sqrt{\frac{2}{\gamma_2 - 1}\left[\left(\frac{P_2}{p_0}\right)^{\frac{\gamma_2 - 1}{\gamma_2}} - 1\right]} \tag{1}$$

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Airspeed. - The following equation was used to calculate airspeed:

$$V_0 = M_0 \sqrt{\gamma g R T_2 \left(\frac{p_0}{P_2}\right)^{\frac{\gamma_2 - 1}{\gamma_2}}} \tag{2}$$

Temperature. - Total temperatures were determined from indicated temperatures by the following relation:

$$T = \frac{T_i \left(\frac{P}{p}\right)^{\frac{\gamma - 1}{\gamma}}}{1 + 0.85\left[\left(\frac{P}{p}\right)^{\frac{\gamma - 1}{\gamma}} - 1\right]} \tag{3}$$

where 0.85 is the impact recovery factor for the type of thermocouple used.

Air flow. - The air flow was determined from pressure and temperature measurements by the following equation:

$$W_{a,1} = p_1 A_1 \sqrt{\frac{2\gamma_1 g}{(\gamma_1 - 1) R t_1}\left[\left(\frac{P_1}{p_1}\right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1\right]} \tag{4}$$

Gas flow. - The gas flow downstream of the combustor was calculated as follows:

$$W_{g,5} = W_{a,1} + \frac{W_f}{3600} \tag{5}$$

Scale thrust. - Values of thrust based on scale measurements were
2733 found for both the data with fixed-exhaust-nozzle areas and control-scheduled data. The jet thrust of the installation was determined from the balance-scale measurements by using the following equation:

$$F_{j,s} = B + D + \frac{W_{a,1}\,V_1}{g} + A_1(p_1 - p_0) \tag{6}$$

When a tail rake was installed, the drag of the rake was added to the right side of the equation. The last two terms of this expression represent the momentum and pressure forces on the installation at the slip joint in the inlet-air duct. The external drag of the installation was determined with the engine inoperative.

Scale net thrust was obtained by subtracting the free-stream momentum of the inlet air from the scale jet thrust:

$$F_{n,s} = F_{j,s} - \frac{W_{a,1}\,V_0}{g} \tag{7}$$

Calculated thrust. - For the data shown in figures 11 through 13, thrust was calculated from conditions at the turbine outlet. For the experimental data, turbine-outlet conditions were measured; while, for data calculated from pumping characteristics, the turbine-outlet conditions were predicted from data at other flight conditions.

Ideal jet thrust was calculated from conditions at the turbine outlet by the following equation:

$$F_{j,r} = \frac{W_{g,6}}{g} V_{eff} \tag{8}$$

In a perfect converging exhaust nozzle,

$$V_{eff} = V_n + \frac{A_n(p_n - p_0)}{\frac{W_{g,6}}{g}} \tag{9}$$

where V_n, A_n, and p_n are the velocity, the area, and the static pressure at the vena contracta. The term $V_{eff}/\sqrt{gRT_6}$ is called the effective velocity parameter and is a function of the exhaust-nozzle pressure ratio and specific heat ratio, as given in figure 14. A further discussion of the effective velocity concept is given in reference 9.

The thrust calculated by equation (8) is an ideal thrust in that it does not include total-pressure losses in the tail pipe and the exhaust nozzle. These losses may most easily be considered by means of an effective velocity coefficient (fig. 10), which is defined as the ratio of scale jet thrust to jet thrust calculated at turbine-outlet conditions. The effective velocity coefficient was obtained from the data given in tables I and II and was found to be primarily a function of turbine-outlet Mach number. Inasmuch as it is impractical to calculate turbine-outlet Mach number by means of a static pressure, a more practical means was used. From continuity considerations 2733

$$\frac{W_{g,6}\sqrt{T_6}}{KP_6} = f(M_6) \tag{10}$$

where K is a constant equal to the effective flow area at the turbine outlet. In the data presented in figure 10, in which effective velocity coefficient C_v is shown as a function of turbine-outlet gas-flow parameter $W_{g,6}\sqrt{T_6}/P_6$ the constant K has been included in the values of the gas-flow parameter on the abscissa.

For the data for which calculated rather than scale values of thrust were used, the exhaust-nozzle pressure ratios p_0/p_6 may be below the limit imposed by the tunnel equipment. However, effective velocity coefficients based on a convergent nozzle are only slightly affected at exhaust-nozzle pressure ratios below critical.

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APPENDIX B

DERIVATION OF COMBUSTION PARAMETER, W_aT_6

If the turbine nozzles are assumed choked,

$$\frac{W_g\sqrt{T_5}}{P_5} = K_1 \tag{11}$$

Experimental results from various engines show that in the range of operation where the turbine nozzles are choked the following relation is valid:

$$T_5 \cong K_2T_6 \tag{12}$$

Combining the two equations yields

$$\frac{W_g\sqrt{T_6}}{P_5} \cong \frac{K_1}{\sqrt{K_2}} \tag{13}$$

Since $W_g \cong W_a$ and $P_5 \cong P_4$

$$\frac{W_a\sqrt{T_6}}{P_4} \cong \frac{K_1}{\sqrt{K_2}} \tag{14}$$

or

$$P_4 \cong \frac{\sqrt{K_2}W_a\sqrt{T_6}}{K_1} \tag{15}$$

Because the Mach numbers are low at the combustor inlet $(M < 0.2)$, the total temperature and pressure can be used with little error in place of the static temperature and pressure so that

$$\rho_4 = \frac{P_4}{gRT_4} \tag{16}$$

and

$$V_4 = \frac{W_aRT_4}{P_4A_4} \tag{17}$$

Substituting equations (15) and (17) for pressure and velocity, respectively, in P_4T_4/V_4 yields the following equation:

$$\frac{P_4T_4}{V_4} \cong \frac{K_2A_4W_aT_6}{K_1{}^2R} \tag{18}$$

The parameter P_4T_4/V_4 has often been used to correlate combustion efficiency. Because all the terms in the right side of equation (17) 2733
are constants except W_aT_6, it may be used in place of P_4T_4/V_4 to correlate combustion efficiency.

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APPENDIX C

SAMPLE PROBLEM

The thrust and the fuel flow are calculated for the conditions of run 54 of table II. The following quantities are known:

$$p_0 = 222 \text{ lb/sq ft} \qquad T_6 = 1532^\circ \text{ R}$$

$$P_2 = 288 \text{ lb/sq ft} \qquad N = 7260 \text{ rpm}$$

$$T_2 = 435^\circ \text{ R}$$

From these quantities the following parameters may be calculated:

$$N/\sqrt{\theta} = 7934 \text{ rpm} \qquad \sqrt{\theta} = 0.915$$

$$T_6/T_2 = 3.50 \qquad \delta/\varphi\sqrt{\theta} = 0.168$$

$$\delta = 0.1361 \qquad V_0 = 610 \text{ ft/sec}$$

$$\theta = 0.838$$

From figures 6(a) and 7(a),

$$\left(\frac{P_6}{P_2}\right)_{\delta/\varphi\sqrt{\theta} = 0.222} = 2.130$$

$$\left(\frac{W_a\sqrt{\theta}}{\delta}\right)_{\delta/\varphi\sqrt{\theta} = 0.222} = 148.2 \text{ lb/sec}$$

From figures 6(b) and 7(b),

Correction factor for pressure ratio = 0.992

Correction factor for corrected air flow = 1.000

Therefore

$$\left(\frac{P_6}{P_2}\right)_{\delta/\varphi\sqrt{\theta} = 0.168} = 2.113$$

$$\left(\frac{W_a\sqrt{\theta}}{\delta}\right)_{\delta/\varphi\sqrt{\theta} = 0.168} = 148.2 \text{ lb/sec}$$

$$(W_a)_{\delta/\varphi\sqrt{\theta}=0.168} = 22.04 \text{ lb/sec}$$

$$(P_6)_{\delta/\varphi\sqrt{\theta}=0.168} = 609 \text{ lb/sq ft}$$

In order to calculate fuel flow and thereby obtain gas flow, the following steps are required:

$$W_aT_6 = (22.04)(1532) = 3.38\times10^4 \text{ (lb)(}^{\circ}\text{R)/sec}$$

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From figure 8,

$$\eta_b = 0.928$$

The engine temperature rise is

$$T_6 - T_2 = 1097^{\circ} \text{ R}$$

From figure 9,

$$(W_f/3600\ W_a)_{ideal} = 0.0152$$

The actual fuel-air ratio is

$$(W_f/3600\ W_a)_{actual} = \frac{0.0152}{0.928} = 0.0164$$

The gas flow is

$$W_{g,6} = W_a\left[1 + (W_f/3600\ W_a)_{actual}\right]$$

$$= (22.04)(1.0164)$$

$$= 22.40 \text{ lb/sec}$$

The next steps in the calculation of thrust are as follows:

$$p_0/P_6 = 222/609$$

$$= 0.365$$

$\gamma = 1.336$ for a $W_f/3600\ W_a$ of 0.0164 and a T_6 of 1532° R

From figure 14,

$$\frac{V_{eff}}{\sqrt{gRT_6}} = 1.328$$

and

$$V_{eff} = 1.328\sqrt{(32.2)(53.4)(1532)}$$

$$= 2155 \text{ ft/sec}$$

The ideal or rake jet thrust is

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$$F_{j,r} = (W_{g,6}/g)\ V_{eff}$$

$$= \frac{22.40}{32.2}\ (2155)$$

$$= 1499 \text{ lb}$$

The inlet momentum is

$$\left(\frac{W_{a,1}}{g}\right) V_0 = \frac{22.04}{32.2}\ (610)$$

$$= 418 \text{ lb}$$

The ideal or rake net thrust is

$$F_{n,r} = F_{j,r} - \frac{W_{a,1}V_0}{g}$$

$$= 1499 - 418$$

$$= 1082 \text{ lb}$$

The fuel flow is

$$W_f = 3600\ W_{a,1}\left[(W_f/3600\ W_{a,1})_{actual}\right]$$

$$= (3600)(22.04)(0.0164)$$

$$= 1301 \text{ lb/hr}$$

Values of calculated ideal net thrust and fuel flow are 1082 pounds and 1301 pounds per hour, respectively. The values from the data are 1087 pounds and 1292 pounds per hour. Therefore, the calculated values are 0.37 percent low for ideal net thrust and 0.70 percent high for fuel flow.

In order to calculate an actual or more realistic thrust, it is necessary to obtain an effective velocity coefficient. The following steps are required:

$$\frac{W_{g,6}\sqrt{T_6}}{P_6} = \frac{22.40\sqrt{1532}}{609} = 1.439$$

Using this value and figure 10, 2733

$$C_v = 0.940$$

The actual jet thrust is

$$(F_j)_{actual} = C_v\ (F_{j,r})$$

$$= (0.940)(1499)$$

$$= 1409 \text{ lb}$$

The actual net thrust is

$$(F_n)_{actual} = (F_j)_{actual} - \frac{W_{a,1}V_0}{g}$$

$$= 1409 - 418$$

$$= 991 \text{ lb}$$

The specific fuel consumption is

$$W_f/F_n = \frac{1301}{991} = 1.313$$

It should be noted that for any engine condition for which the performance may be desired, the corresponding engine speed and exhaust-gas temperature must be within the physical capabilities of the exhaust nozzle. This can be verified by the data of figure 5.

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TABLE I. - FIXED-

Run	Altitude (ft)	Ram pressure ratio P_2/p_0	Flight Mach number M_0	Tunnel static pressure p_0 (lb/sq ft abs)	Reynolds number index $\delta_2/\varphi_2\sqrt{\theta_2}$	Equivalent ambient-air temperature $t_{0,e}$ (°R)	Engine-inlet total temperature T_2 (°R)	Actual engine speed N (rpm)	Corrected engine speed $N/\sqrt{\theta_2}$ (rpm)	Ideal net thrust $F_{n,r}$ (lb)	Actual net thrust $F_{n,s}$ (lb)	Corrected net thrust $F_{n,s}/\delta_2$ (lb)	Ideal jet thrust $F_{j,r}$ (lb)
													Exhaust-nozzle
1	15,000	1.298	0.622	1187	0.758	464	500	6534	6658	4001	3749	5151	5882
2		1.301	.625	1184	.756	465	501	6353	6467	3511	3255	4472	5315
3		1.296	.621	1183	.754	464	500	6171	6288	3059	2827	3901	4762
4		1.298	.622	1186	.757	464	500	5808	5918	2296	2036	2800	3845
5		1.288	.613	1190	.754	463	500	5082	5179	1100	997	1377	2322
6	35,000	1.303	0.627	479	0.358	413	445	6171	6665	1691	1573	5332	2439
7		1.303	.627	479	.358	413	446	5990	6483	1487	1363	4621	2201
8		1.304	.628	477	.355	413	446	5808	6267	1279	1216	4142	1952
9		1.299	.623	478	.354	414	446	5445	5875	925	855	2914	1521
10		1.296	.621	479	.354	414	446	5082	5483	856	588	1935	1195
11		1.873	.992	476	.482	390	467	6534	6887	2677	2436	5781	4414
12		1.873	.992	473	.484	386	463	6353	6728	2368	----	----	4019
13		1.869	.990	479	.488	389	465	6171	6517	2051	----	----	3639
14		1.872	.991	478	.479	393	470	5808	6104	1457	1313	3104	2898
15		1.863	.987	481	.480	393	469	5082	5546	615	493	1164	1802
16	45,000	1.295	0.619	288	0.221	404	435	5990	6541	983	884	5015	1377
17		1.327	.649	286	.224	402	436	5808	6337	840	793	4420	1257
18		1.311	.634	288	.222	404	437	5445	5935	610	567	3176	978
19		1.313	.636	289	.222	405	438	5082	5534	428	371	2069	761
													Exhaust-nozzle
20	15,000	1.292	0.616	1186	0.747	468	504	7260	7369	4645	4382	6052	6781
21		1.291	.616	1184	.746	468	503	7079	7192	4304	4114	5694	6386
22		1.292	.616	1188	.748	467	503	6897	7007	3946	3757	5181	5974
23		----	----	1188	----	---	504	6716	6817	----	----	----	----
24		1.295	.619	1183	.747	467	505	6534	6659	3139	2988	4786	5029
25		1.295	.619	1179	.746	467	503	6171	6270	2363	2264	3136	4085
26		1.291	.616	1185	.742	469	505	5808	5889	1709	1613	2251	3261
27		1.295	.619	1184	.744	469	505	5082	5153	751	662	914	2015
28	35,000	1.303	0.627	479	0.360	411	443	7260	7863	2265	2133	7229	3180
29		1.297	.621	478	.358	412	444	7079	7652	2160	2051	6984	3056
30		1.297	.621	478	.358	412	444	6897	7456	1989	1951	6657	2852
31		1.295	.619	479	.358	412	444	6716	7260	1876	1783	6084	2731
32		1.302	.626	479	.359	412	444	6534	7063	1715	1575	5346	2561
33		1.300	.624	479	.358	413	445	6171	6665	1333	1271	4319	2106
34		1.298	.622	478	.357	413	445	5808	6273	973	905	3087	1667
35		1.302	.626	479	.358	413	445	5082	5489	489	407	1381	1017
36		1.852	.982	481	.479	394	470	7260	7830	2909	2724	8472	4835
37		1.867	.989	479	.482	391	467	7079	7461	2783	2580	6104	4703
38		1.850	.981	478	.478	392	467	6897	7269	2542	2392	5724	4395
39		1.865	.988	479	.478	395	472	6716	7045	2291	2124	5030	4106
40		1.857	.984	481	.471	400	478	6534	6808	1997	1928	4567	3736
41		1.861	.986	478	.478	394	471	6171	6480	1522	1378	3278	3144
42		1.863	.987	480	.480	393	469	5808	6110	1049	875	2071	2535
43		1.850	.981	481	.479	394	470	5082	5341	310	205	487	1502
44	45,000	1.287	0.612	290	0.222	407	437	7079	7716	1385	1319	7479	1914
45		1.280	.614	291	.225	406	437	6897	7518	1337	1256	7078	1868
46		1.288	.613	289	.222	406	436	6716	7327	1357	1138	6470	1924
47		1.299	.623	287	.222	405	436	6534	7129	1112	1047	5943	1619
48		1.291	.616	289	.222	405	436	6171	6733	875	801	4542	1334
49		1.301	.625	288	.224	403	435	5808	6342	635	601	3393	1055
50		1.289	.614	287	.223	405	435	5082	5550	314	274	1868	840
													Exhaust-nozzle
51	15,000	1.294	0.619	1186	0.755	466	502	7260	7383	4076	3870	5057	6216
52		1.288	.614	1186	.752	465	500	7079	7214	3904	3580	4927	5996
53		1.289	.614	1191	.760	460	495	6897	7063	3511	3178	4382	5530
54		1.288	.613	1194	.744	471	506	6716	6803	3110	2839	3906	5084
55		1.291	.616	1188	.751	467	502	6534	6645	2822	2583	3562	4728
56		1.300	.624	1182	.753	466	502	6171	6276	2035	1832	2524	3785
57		1.295	.619	1187	.735	475	511	5808	5854	1429	1219	1679	2999
58		1.301	.625	1187	.743	470	507	5082	5143	526	406	557	1812
59	35,000	1.288	0.622	480	0.358	414	446	7260	7834	2020	1815	6167	2926
60		1.286	.611	484	.355	419	450	7079	7603	1924	1761	5887	2803
61		1.299	.623	479	.353	419	452	6897	7387	1784	1647	5600	2854
62		1.303	.627	480	.356	418	451	6716	7206	1658	1451	4910	2523
63		1.292	.616	481	.353	421	453	6534	6998	1484	1305	4444	2305
64		1.294	.619	480	.353	421	453	6171	6609	1152	1026	3495	1909
65		1.303	.627	479	.352	421	454	5808	6209	824	757	2498	1519
66		1.297	.621	479	.350	422	455	5082	5428	661	294	1001	908
67		1.865	.988	477	.479	393	470	7260	7630	2551	2327	5534	4494
68		1.857	.984	478	.478	394	470	7074	7440	2356	2234	5526	4227
69		1.870	.990	477	.477	395	472	6897	7235	2648	----	----	4902
70		1.853	.982	477	.473	396	472	6716	7045	2023	1827	4374	3834
71		1.873	.992	479	.478	394	472	6534	6854	1787	1580	3726	3572
72		1.873	.992	478	.477	394	471	6171	6480	1309	1134	2680	2951
73		1.868	.989	478	.477	395	472	5808	6093	849	708	1678	2336
74		1.854	.983	479	.476	395	471	5082	5336	197	107	255	1392
75	45,000	1.289	0.614	280	0.213	418	447	7260	7819	1289	1216	6884	1828
76		1.296	.621	289	.213	415	447	7079	7624	1240	1140	6440	1780
77		1.288	.613	290	.214	413	444	6897	7456	1161	1083	6024	1685
78		1.283	.608	289	.212	414	445	6716	7253	1078	963	5496	1586
79		1.303	.627	280	.215	409	441	6534	7089	982	903	5055	1495
80		1.288	.613	289	.216	411	442	6171	6688	776	664	3775	1238
81		1.299	.623	287	.216	410	442	5808	6296	562	510	2895	980
82		1.311	.634	288	.217	407	440	5082	5519	260	224	1255	598

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EXHAUST-NOZZLE DATA

Actual jet thrust $F_{j,s}$ (lb)	Corrected jet thrust $F_{j,s}/\delta_2$ (lb)	Actual air flow $W_{a,1}$ (lb/sec)	Corrected air flow $W_{a,1}\sqrt{\theta_2}/\delta_2$ (lb/sec)	Actual fuel flow W_f (lb/hr)	Corrected fuel flow $W_f/\delta_2\sqrt{\theta_2}$ (lb/hr)	Actual specific fuel consumption $W_f/F_{n,s}$ (lb)/(hr)(lb thrust)	Corrected specific fuel consumption $W_f/F_{n,s}\sqrt{\theta_2}$ (lb)/(hr)(lb thrust)	Actual exhaust-gas temperature T_6 (°R)	Corrected exhaust-gas temperature T_6/θ_2 (°R)	Engine total-pressure ratio P_6/P_2	Engine total-temperature ratio T_6/T_2	Run
area, 367 sq in.												
5630	7,736	92.15	124.31	4975	6,966	1.327	1.352	1529	1587	1.870	3.058	1
5059	6,951	87.82	118.56	4405	6,161	1.353	1.376	1472	1525	1.749	2.938	2
4530	6,251	83.62	113.22	3910	5,498	1.383	1.409	1408	1462	1.644	2.816	3
3585	4,929	75.89	102.45	3095	4,337	1.520	1.549	1302	1351	1.464	2.604	4
2219	3,064	60.71	82.26	1884	2,652	1.890	1.926	1111	1153	1.176	2.222	5
2321	7,868	38.51	120.88	2070	7,580	1.316	1.421	1476	1721	1.916	3.317	6
2077	7,041	36.78	115.60	1861	6,807	1.365	1.473	1416	1648	1.785	3.175	7
1891	6,431	34.60	109.09	1659	6,068	1.362	1.470	1354	1576	1.676	3.036	8
1451	4,945	30.85	97.46	1300	4,781	1.520	1.641	1242	1446	1.466	2.785	9
1107	3,772	28.00	88.42	1032	3,795	1.817	1.961	1133	1319	1.300	2.540	10
4173	9,969	58.23	131.08	3205	8,016	1.316	1.387	1547	1719	1.995	3.313	11
----	-----	55.65	125.55	2840	7,186	-----	-----	1464	1641	1.878	3.162	12
----	-----	53.38	119.46	2470	6,165	-----	-----	1391	1551	1.731	2.991	13
2754	6,510	48.14	108.32	1850	4,596	1.409	1.481	1227	1356	1.478	2.611	14
1680	3,966	39.82	89.36	991	2,463	2.010	2.116	951	1053	1.088	2.028	15
1298	7,364	21.81	113.28	1315	8,146	1.488	1.624	1525	1818	1.869	3.506	16
1210	6,745	21.01	107.34	1173	7,135	1.479	1.614	1435	1708	1.733	3.291	17
935	5,238	18.96	97.45	942	5,753	1.661	1.811	1303	1548	1.496	2.982	18
704	3,926	17.06	87.38	779	4,728	2.100	2.286	1180	1399	1.325	2.694	19
area 421 sq in.												
6518	9,001	105.12	143.07	5530	7,752	1.262	1.281	1527	1573	1.935	3.030	20
6196	8,575	102.58	139.82	5150	7,241	1.252	1.272	1474	1521	1.872	2.930	21
5785	7,978	99.94	135.72	4710	6,599	1.254	1.274	1424	1470	1.791	2.831	22
----	-----	-----	-----	4345	-----	-----	-----	-----	----	-----	-----	23
4878	6,737	92.70	126.07	3860	5,416	1.292	1.313	1321	1363	1.620	2.626	24
3986	5,521	84.44	115.18	3080	4,334	1.360	1.382	1219	1258	1.454	2.423	25
3165	4,377	76.41	104.22	2440	3,422	1.513	1.534	1133	1165	1.304	2.244	26
1926	2,658	61.90	84.25	1693	2,229	2.406	2.440	990	1018	1.063	1.960	27
3048	10,330	47.24	147.91	2605	9,560	1.221	1.323	1530	1795	2.114	3.454	28
2947	10,035	46.65	146.90	2450	9,016	1.195	1.291	1483	1734	2.069	3.340	29
2814	9,601	44.94	141.83	2275	8,390	1.166	1.260	1428	1669	2.009	3.216	30
2639	9.004	44.69	141.04	2115	7,800	1.186	1.282	1383	1617	1.935	3.115	31
2421	8,217	43.68	137.11	1935	7,100	1.229	1.328	1331	1556	1.851	2.998	32
2044	6,946	40.03	125.97	1580	5,797	1.243	1.342	1220	1423	1.654	2.742	33
1599	5,454	36.04	113.86	1250	4,606	1.381	1.492	1122	1308	1.454	2.521	34
965	3,275	28.79	90.49	855	3,133	2.101	2.268	979	1142	1.164	2.200	35
4650	11,048	64.86	146.69	3425	8,554	1.257	1.322	1507	1665	2.009	3.206	36
4500	10,647	64.47	144.67	3205	7,992	1.242	1.309	1467	1630	1.953	3.141	37
4245	10,158	62.64	142.19	2950	7,440	1.233	1.300	1410	1567	1.889	3.019	38
3939	9,328	60.70	137.06	2670	6,633	1.267	1.319	1358	1494	1.780	2.877	39
3667	8,687	57.99	131.87	2395	5,913	1.242	1.295	1301	1413	1.659	2.722	40
3000	7,157	54.38	123.23	1855	4,634	1.346	1.414	1147	1265	1.486	2.435	41
2361	5,588	49.84	112.14	1413	3,517	1.615	1.698	1030	1140	1.267	2.196	42
1397	3,322	40.19	90.95	767	1,917	3.741	3.832	815	901	.959	1.734	43
1850	10,490	28.24	146.93	1650	10,200	1.251	1.364	1551	1843	2.146	3.549	44
1787	10,070	28.13	145.46	1570	9,641	1.250	1.362	1515	1800	2.101	3.467	45
1705	9,693	------	------	1440	8,931	1.265	1.380	1452	1728	2.037	3.330	46
1554	8,821	26.55	138.01	1319	8,168	1.260	1.374	1381	1643	1.937	3.167	47
1260	7,144	24.30	126.29	1095	6,776	1.367	1.492	1278	1521	1.728	2.931	48
1021	5,765	21.92	113.30	895	5,516	1.489	1.636	1171	1396	1.502	2.692	49
600	3,453	17.33	90.77	683	4,269	2.493	2.723	1029	1227	1.206	2.366	50
area, 449 sq in.												
5810	8,006	105.21	142.56	4680	6,559	1.275	1.297	1393	1440	1.754	2.775	51
5652	7,822	103.79	140.95	4470	6,304	1.256	1.279	1366	1418	1.723	2.732	52
5197	7,167	100.66	135.59	4045	5,712	1.273	1.303	1321	1366	1.630	2.669	53
4793	6,595	96.43	131.05	3700	5,157	1.303	1.320	1270	1303	1.567	2.510	54
4489	6,190	94.06	127.55	3410	4,782	1.320	1.343	1222	1264	1.508	2.434	55
3582	4,936	85.29	115.57	2660	3,727	1.452	1.477	1129	1167	1.342	2.249	56
2789	3,840	76.37	104.32	2140	2,970	1.756	1.769	1073	1090	1.215	2.100	57
1692	2,320	62.26	84.36	1360	1,886	3.350	3.389	934	956	1.018	1.842	58
2721	9,248	46.97	147.96	2310	8,468	1.273	1.373	1399	1628	1.932	3.137	59
2640	8,976	46.18	146.11	2115	7,725	1.201	1.290	1374	1584	1.888	3.053	60
2517	8,558	44.73	141.93	1990	7,245	1.208	1.294	1340	1537	1.828	2.965	61
2316	7,837	44.32	139.83	1825	6,626	1.258	1.349	1280	1473	1.764	2.838	62
2126	7,239	42.59	135.48	1681	6,129	1.288	1.379	1240	1422	1.675	2.737	63
1783	6,073	39.20	124.73	1385	5,051	1.350	1.445	1142	1310	1.510	2.521	64
1432	4,854	35.50	112.57	1105	4,004	1.499	1.602	1055	1204	1.336	2.319	65
841	2,864	28.13	89.71	753	2,738	2.561	2.735	916	1045	1.099	2.013	66
4270	10,154	65.11	147.34	2940	7,348	1.263	1.328	1371	1515	1.816	2.917	67
4105	9,786	62.90	142.72	2740	6,866	1.226	1.289	1323	1462	1.784	2.815	68
----	------	------	------	2515	6,257	-----	-----	1283	1411	1.705	2.718	69
3638	8,709	60.86	138.94	2300	5,777	1.259	1.321	1239	1363	1.633	2.625	70
3365	7,935	59.52	133.86	2050	5,070	1.297	1.361	1171	1288	1.537	2.481	71
2776	6,560	54.81	123.38	1570	3,897	1.384	1.454	1054	1165	1.346	2.242	72
2195	5,202	49.65	112.21	1200	2,984	1.695	1.778	937	1031	1.171	1.985	73
1302	3,101	40.17	91.15	680	1,701	6.358	6.673	753	831	.889	1.599	74
1755	9,935	28.26	148.48	1520	9,267	1.260	1.346	1471	1706	1.993	3.291	75
1680	9,490	28.04	147.01	1450	8,824	1.272	1.370	1430	1659	1.965	3.199	76
1587	8,994	27.64	144.86	1350	8,268	1.270	1.373	1371	1603	1.899	3.088	77
1471	8,395	26.96	142.48	1245	7,676	1.293	1.397	1325	1546	1.837	2.978	78
1416	7,927	26.56	137.05	1152	6,998	1.276	1.384	1267	1491	1.741	2.873	79
1126	6,401	24.42	128.11	950	5,856	1.431	1.551	1174	1379	1.580	2.658	80
928	5,268	21.72	113.79	794	4,888	1.557	1.688	1100	1293	1.390	2.489	81
562	3,149	17.35	89.53	627	3,816	2.799	3.040	953	1124	1.126	2.166	82

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TABLE I. - Concluded.

Run	Altitude (ft)	Ram pressure ratio P_2/p_0	Flight Mach number M_0	Tunnel static pressure p_0 (lb/sq ft abs)	Reynolds number index $\delta_2/\varphi_2\sqrt{\theta_2}$	Equivalent ambient-air temperature $t_{0,e}$ (°R)	Engine-inlet total temperature T_2 (°R)	Actual engine speed N (rpm)	Corrected engine speed $N\sqrt{\theta_2}$ (rpm)	Ideal net thrust $F_{n,r}$ (lb)	Actual net thrust $F_{n,s}$ (lb)	Corrected net thrust $F_{n,s}/\delta$ (lb)	Ideal jet thrust $F_{j,r}$ (lb)
													Exhaust-nozzle
83	15,000	1.298	0.622	1183	0.751	465	501	7260	7391	3732	3411	4700	5905
84		1.292	.616	1186	.750	465	500	7079	7214	2442	3100	4278	5545
85		1.296	.621	1188	.748	468	504	6897	7000	3125	2806	3855	5195
86		1.295	.619	1187	.747	469	505	6716	6810	2789	2498	3440	4781
87		1.300	.624	1187	.749	468	504	6534	6632	2459	2178	2986	4404
88		1.301	.625	1188	.751	468	505	6171	6257	1734	1529	2093	3506
89		1.294	.619	1184	.743	470	506	5808	5884	1217	1061	1465	2802
90		1.295	.619	1186	.740	473	509	5082	5133	429	320	441	1698
91	35,000	1.307	0.631	474	0.352	411	444	7260	7848	1817	1629	5566	2732
92		1.294	.619	478	.355	412	443	7079	7667	1710	1540	5268	2598
93		1.275	.600	483	.350	420	450	7079	7603	1692	1547	5315	2546
94		1.304	.628	480	.361	411	443	6897	7469	1622	1396	4720	2525
95		1.300	.624	477	.354	417	449	6716	7220	1469	1313	4479	2335
96		1.282	.607	479	.350	418	449	6534	7024	1351	1174	4046	2142
97		1.291	.616	482	.356	417	449	6171	6634	1022	887	3016	1792
98		1.302	.626	480	.359	414	446	5808	6267	761	626	2126	1485
99		1.310	.634	479	.359	414	447	5082	5473	323	242	816	907
100		1.879	.984	477	.482	392	469	7260	7638	2311	2079	4909	4280
101		1.870	.990	479	.478	394	471	7079	7433	2183	1971	4656	4124
102		1.882	.996	477	.480	392	470	6897	7249	2010	1768	4167	3920
103		1.850	.981	479	.473	394	470	6716	7059	1795	1561	3702	3605
104		1.865	.988	477	.480	392	468	6534	6880	1599	1558	3706	3376
105		1.857	.984	478	.478	395	471	6171	6480	1129	948	2259	2759
106		1.868	.989	477	.476	394	471	5808	6098	691	504	1197	2183
107		1.854	.983	478	.477	394	470	5062	5341	96	7	17	1303
108	45,000	1.288	0.613	289	0.210	408	439	7260	7892	1183	1047	5952	1721
109		1.300	.624	291	.214	405	437	7079	7716	1139	1029	5755	1691
110		1.301	.625	289	.214	404	436	6897	7525	1084	944	5314	1627
111		1.285	.610	288	.209	420	451	6716	7206	931	798	4561	1435
112		1.289	.614	291	.213	415	446	6534	7050	855	710	4004	1355
113		1.296	.621	291	.218	410	442	6171	6689	690	585	3282	1162
114		1.296	.621	291	.219	409	440	5808	6307	503	447	2508	925
115		1.300	.624	291	.216	415	447	5082	5473	228	140	783	574
													Exhaust-nozzle
116	15,000	1.297	0.621	1181	0.743	467	503	7260	7376	3212	2656	3670	5369
117		1.294	.619	1183	.751	465	500	7079	7190	2955	2463	3406	5059
118		1.295	.619	1183	.741	469	505	6897	6984	----	2167	2993	----
119		1.296	.619	1183	.739	470	506	6716	6803	2323	1905	2633	4303
120		1.295	.619	1183	.738	472	508	6534	6606	2013	1623	2241	3926
121		1.298	.622	1182	.735	472	509	6171	6233	1443	1158	1597	3226
122		1.296	.621	1182	.732	474	510	5808	5860	901	725	1001	2502
123		1.297	.621	1181	.730	474	511	5082	5123	232	138	191	1525
124	35,000	1.293	0.618	479	0.360	409	440	7260	7884	----	1347	4601	----
125		1.295	.619	480	.360	409	440	7079	7688	1539	1274	4337	2435
126		1.293	.618	479	.362	408	439	6897	7497	----	1180	4039	----
127		1.297	.621	479	.359	409	441	6716	7287	1323	1083	3689	2194
128		1.293	.618	481	.369	410	441	6534	7089	----	964	3280	----
129		1.294	.619	481	.359	410	441	6171	6698	952	751	2553	1747
130		1.294	.619	478	.358	411	442	5808	6296	614	480	1643	1328
131		1.298	.622	479	.358	410	442	5082	5509	228	159	541	794
132		1.860	.985	479	.480	392	468	7260	7645	2074	1718	4073	4021
133		1.861	.986	478	.479	392	468	7079	7454	1956	1560	3687	3882
134		1.872	.991	477	.482	391	468	6897	7263	1801	1416	3356	3704
135		1.883	.996	476	.481	391	469	6716	7065	----	1228	2899	----
136		1.874	.992	477	.476	394	472	6534	6854	1368	1045	2470	3159
137		1.884	.997	477	.480	394	472	6171	6473	908	662	1559	2565
138		1.880	.995	476	.478	393	471	5808	6098	536	334	790	2054
139		1.876	.993	478	.479	393	471	5082	5356	-17	-143	-337	1222
140	45,000	1.287	0.612	290	0.210	424	456	7260	7746	992	828	4696	1526
141		1.293	.618	290	.212	420	452	7079	7582	----	785	4485	----
142		1.302	.626	287	.211	417	450	6897	7407	906	697	3946	1443
143		1.294	.619	289	.212	415	447	6716	7233	837	707	3972	1355
144		1.298	.622	290	.215	413	445	6534	7057	768	622	3498	1282
145		1.278	.603	289	.212	414	444	6171	6671	596	492	2819	1054
146		1.285	.619	291	.216	411	443	5808	6290	383	332	1864	807
147		1.300	.624	294	.221	411	443	5082	5504	174	90	498	523
148	50,000	1.284	0.609	224	0.168	405	435	7260	7928	849	716	5266	1266
149		1.266	.591	234	.171	405	433	7079	7752	819	693	4951	1231
150		1.280	.605	225	.150	446	479	6716	6991	586	473	3474	965
151		1.300	.624	222	.152	440	474	6534	6835	499	435	3189	872
152		1.283	.608	226	.147	451	484	5808	6011	250	183	1335	547
153		1.296	.619	225	.147	451	486	5082	5250	84	2	15	323
154	55,000	1.314	0.637	162	0.122	408	441	7079	7681	660	555	5516	984
155		1.315	.638	164	.128	405	438	6716	7314	523	417	4092	838
156		1.304	.628	169	.130	406	438	6534	7118	482	388	3696	785
157		1.295	.619	168	.129	407	438	5808	6325	262	240	2334	507
158		1.279	.604	170	.124	410	440	5082	5519	124	91	885	309

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FIXED-EXHAUST-NOZZLE DATA

Actual jet thrust $F_{j,s}$ (lb)	Corrected jet thrust $F_{j,s}/\delta_2$ (lb)	Actual air flow $W_{a,1}$ (lb/sec)	Corrected air flow $W_{a,1}\sqrt{\theta_2}/\delta_2$ (lb/sec)	Actual fuel flow W_f (lb/hr)	Corrected fuel flow $W_f/\delta_2\sqrt{\theta_2}$ (lb/hr)	Actual specific fuel consumption $W_f/F_{n,s}$ (lb)/(hr)(lb thrust)	Corrected specific fuel consumption $W_f/F_{n,s}\sqrt{\theta_2}$ (lb)/(hr)(lb thrust)	Actual exhaust-gas temperature T_6 (°R)	Corrected exhaust-gas temperature T_6/θ_2 (°R)	Engine total-pressure ratio P_6/P_2	Engine total-temperature ratio T_6/T_2	Run
area, 479 sq in.												
5584	7,695	106.33	143.97	4370	6,131	1.281	1.304	1308	1355	1.659	2.611	83
5203	7,180	103.84	140.60	4030	5,668	1.300	1.325	1259	1307	1.615	2.518	84
4876	6.700	101.22	137.05	3715	5,181	1.324	1.344	1224	1261	1.547	2.429	85
4490	6,163	97.50	132.41	3385	4,698	1.347	1.366	1189	1222	1.481	2.354	86
4123	5,653	94.58	127.78	3035	4,224	1.393	1.415	1145	1179	1.412	2.272	87
3301	4,519	85.96	116.05	2395	3,325	1.566	1.589	1060	1090	1.265	2.099	88
2646	3,654	77.58	105.82	1945	2,721	1.833	1.457	1005	1031	1.162	1.986	89
1589	2,190	61.83	84.40	1287	1,791	4.022	4.063	908	926	.998	1.784	90
2544	8,893	46.96	148.59	2000	7.388	1.228	1.327	1292	1510	1.812	2.910	91
2428	8,306	46.45	146.83	1882	6,972	1.222	1.325	1248	1464	1.757	2.817	92
2403	8,257	45.71	146.27	1855	6,845	1.199	1.288	1256	1448	1.764	2.791	93
2299	7,773	46.56	145.45	1770	6,481	1.268	1.373	1203	1411	1.708	2.716	94
2179	7,433	44.63	141.61	1620	5,942	1.234	1.327	1172	1355	1.648	2.610	95
1985	6,840	42.88	137.43	1512	5,600	1.288	1.384	1138	1316	1.591	2.535	96
1657	5,634	40.14	126.92	1245	4,549	1.404	1.508	1047	1210	1.428	2.332	97
1352	4,578	37.31	117.12	1030	3,762	1.640	1.769	978	1138	1.286	2.193	98
826	2,786	29.72	93.02	750	2,725	3.099	3.339	864	1002	1.068	1.933	99
4048	9,857	65.69	147.41	2615	6,495	1.258	1.323	1276	1413	1.701	2.721	100
3912	9,240	64.85	145.91	2475	6,139	1.256	1.319	1246	1344	1.649	2.645	101
3678	8,669	63.58	142.61	2255	5,586	1.275	1.340	1194	1319	1.596	2.540	102
3361	8.023	61.02	138.58	2040	5,118	1.315	1.382	1148	1269	1.531	2.443	103
3335	7,934	59.65	134.75	1826	4,572	1.171	1.234	1097	1217	1.450	2.344	104
2578	6,143	54.71	124.19	1435	3,591	1.514	1.590	988	1090	1.274	2.098	105
1996	4,741	49.89	112.85	1049	2,615	2.081	2.185	880	971	1.095	1.868	106
1214	2,899	40.61	92.27	800	1,507	85.71	90.14	723	799	.849	1.538	107
1585	9,011	28.52	149.10	1374	8,493	1.312	1.427	1368	1617	1.865	3.116	108
1581	8,843	28.82	147.90	1310	7,987	1.275	1.388	1315	1562	1.821	3.009	109
1487	8,370	28.33	146.15	1223	7,509	1.296	1.413	1281	1524	1.790	2.938	110
1302	7,442	26.45	140.93	1101	6,751	1.380	1.480	1239	1426	1.688	2.747	111
1210	6,824	26.23	137.13	1020	6,210	1.437	1.551	1185	1377	1.616	2.652	112
1057	5,930	24.65	127.61	887	5,397	1.516	1.644	1105	1296	1.477	2.495	113
869	4,875	22.08	114.07	773	4,707	1.729	1.877	1038	1224	1.320	2.369	114
486	2,719	17.88	92.85	663	3,995	4.736	5.100	942	1093	1.079	2.107	115
area, 535 sq in.												
4813	6,652	105.44	143.50	3760	5,279	1.416	1.438	1224	1263	1.529	2.433	116
4567	6,316	103.46	140.71	3495	4,910	1.419	1.442	1182	1223	1.483	2.370	117
4215	5,821	100.26	136.54	3200	4,481	1.477	1.497	1145	1177	-----	2.267	118
3885	5,369	96.80	132.13	2885	4,040	1.514	1.534	1107	1136	1.367	2.188	119
3536	4,883	93.31	127.46	2625	3,665	1.617	1.635	1088	1091	1.310	2.102	120
2941	4,056	86.62	118.32	----	-----	-----	-----	1003	1023	1.194	1.971	121
2326	3,212	77.76	106.45	1710	2,382	2.359	2.379	936	953	1.090	1.835	122
1431	1,976	62.73	85.94	1170	1,628	6.478	8.543	843	856	.956	1.650	123
2245	7,669	47.20	148.44	1802	6,685	1.338	1.453	1204	1420	-----	2.736	124
2170	7,387	46.98	147.24	1690	6,246	1.327	1.440	1174	1384	1.628	2.668	125
2059	7,048	46.24	145.56	1580	5,877	1.339	1.455	1128	1333	-----	2.569	126
1954	6,655	45.51	142.90	1480	5,470	1.367	1.483	1095	1289	1.536	2.485	127
1805	6,142	44.16	138.53	1360	5,023	1.411	1.531	1046	1231	-----	2.372	128
1546	5,256	41.68	130.63	1165	4,298	1.551	1.683	992	1168	1.362	2.249	129
1194	4,086	37.34	117.92	915	3,395	1.906	2.067	902	1060	1.205	2.041	130
725	2,467	29.49	92.60	680	2,508	4.277	4.635	805	946	1.018	1.821	131
3662	8,697	66.51	147.73	2330	5,826	1.359	1.430	1205	1336	1.581	2.575	132
3476	8,269	64.76	146.29	2190	5,486	1.413	1.488	1164	1291	1.550	2.487	133
3319	7,866	63.77	143.56	2030	5,067	1.434	1.510	1122	1244	1.497	2.397	134
3089	7,293	62.03	139.20	1815	4,507	1.478	1.555	1055	1166	-----	2.249	135
2834	6,711	59.70	134.90	1640	4,073	1.572	1.649	1021	1123	1.347	2.163	136
2319	5,461	55.00	125.53	1245	3,076	1.881	1.973	912	1003	1.172	1.932	137
1852	4,378	50.51	113.75	938	2,329	2.808	2.949	818	902	1.030	1.737	138
1096	2,587	41.30	92.84	528	1,307	-----	-----	685	756	.800	1.454	139
1362	7,724	27.83	147.92	1200	7,259	1.449	1.546	1265	1440	1.686	2.774	140
1332	7,514	27.86	146.66	1130	6,826	1.421	1.522	1254	1415	-----	2.730	141
1234	6,986	27.59	145.45	1057	6,425	1.516	1.628	1193	1376	1.621	2.651	142
1220	6,903	26.98	141.67	1000	6,094	1,425	1.534	1158	1343	1.564	2.591	143
1136	6,388	26.67	138.87	930	5,646	1.495	1.614	1109	1293	1.512	2.492	144
950	5,444	24.51	129.90	789	4,888	1.604	1.734	1031	1205	1.389	2.322	145
756	4,244	22.19	115.10	673	4,093	2.027	2.194	937	1099	1.209	2.115	146
439	2,431	18.10	92.62	585	3.511	6.500	7.044	871	1022	1.034	1.966	147
1133	8,333	22.36	150.57	995	7,995	1.390	1.518	1290	1538	1.756	2.966	148
1105	7,894	22.76	148.51	953	7,458	1.375	1.506	1234	1480	1.712	2.850	149
852	6,256	19.45	137.24	801	6,126	1.693	1.783	1214	1316	1.486	2.534	150
808	5,924	18.73	131.24	750	5,756	1.724	1.809	1147	1255	1.407	2.420	151
480	3,503	15.11	106.48	659	4,977	3.601	3.727	1027	1100	1.159	2.122	152
241	1,751	11.92	83.79	566	4,264	284.0	293.5	931	993	.982	1.916	153
879	8,736	16.53	151.45	654	9.213	1.539	1.670	1337	1574	1.815	3.032	154
732	7,184	16.12	145.32	779	8,322	1.868	2.034	1138	1350	1.647	2.698	155
688	6,605	15.69	138.37	744	7,777	1.932	2.104	1126	1335	1.570	2.571	156
485	4,716	12.85	114.78	855	6,933	2.729	2.971	977	1159	1.273	2.231	157
276	2,686	9.96	89.22	637	6,732	7.000	7.604	924	1089	1.071	2.100	158

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TABLE II. - CONTROL DATA

Run	Altitude (ft)	Ram pressure ratio P_2/p_0	Flight Mach number M_0	Tunnel static pressure p_0 (lb/sq ft abs)	Reynolds number index $\delta_2/\varphi_2\sqrt{\theta_2}$	Equivalent ambient-air temperature $t_{0,e}$ (°R)	Engine-inlet total temperature T_2 (°R)	Engine speed N (rpm)	Exhaust-nozzle projected area A_n (sq in.)	Ideal net thrust $F_{n,r}$ (lb)	Actual net thrust $F_{n,s}$ (lb)	Ideal jet thrust $F_{j,r}$ (lb)	Actual jet thrust $F_{j,s}$ (lb)	Air flow $W_{a,1}$ (lb/sec)	Fuel flow W_f (lb/hr)	Specific fuel consumption $W_f/F_{n,s}$ (lb)/(hr)(lb thrust)	Exhaust-gas temperature T_6 (°R)	Engine total-pressure ratio P_6/P_2	Engine total-temperature ratio T_6/T_2
1	15,000	1.302	0.626	1163	0.757	464	500	7260	418	4768	4346	6947	6528	106.21	5555	1.278	1532	1.933	3.064
2		1.291	.618	1168	.763	461	498	7260	442	4358	3858	6606	6108	106.81	4865	1.259	1425	1.828	2.875
3		1.292	.616	1163	.752	465	500	7260	486	4014	3642	6151	5779	105.51	4615	1.267	1366	1.747	2.732
4		1.298	.621	1163	.754	465	501	7260	511	3369	2847	5508	5097	106.50	3806	1.325	1257	1.676	2.469
5		1.298	.629	1163	.739	472	508	6718	536	2298	1837	4273	3812	98.92	2895	1.578	1129	1.386	2.209
6		1.292	.616	1168	.739	473	509	6554	535	1983	1607	3879	3503	92.82	2605	1.621	1077	1.298	2.116
7		1.299	.623	1168	.749	469	508	5808	536	----	689	----	2256	77.84	1680	2.549	941	1.063	1.865
8		1.291	.616	1190	.731	477	513	5082	536	216	167	1471	1422	61.25	1160	6.946	874	.948	1.704
9		1.284	.609	1190	.739	472	507	3993	536	-195	-253	700	640	44.30	775	-----	790	.868	1.558
10		1.287	.612	1164	.741	468	503	5086	536	-350	-336	329	323	32.70	584	-----	682	.912	1.358
11	35,000	1.017	0.188	477	0.278	443	445	7280	435	2044	1886	2225	2067	36.48	2070	1.098	1585	2.110	3.617
12		1.020	.189	478	.279	441	444	7280	435	2040	1889	2239	2088	36.84	2071	1.096	1558	2.101	3.509
13		1.014	.141	478	.278	442	444	7280	475	1810	1681	1978	1844	36.46	1760	1.041	1408	1.918	3.171
14		1.018	.160	478	.281	441	443	7260	536	1560	1398	1748	1584	36.78	1510	1.082	1281	1.734	2.847
15		1.018	.160	478	.290	440	442	7079	535	1498	1342	1683	1528	36.57	1460	1.080	1217	1.705	2.785
16		1.019	.164	478	.285	458	440	6718	536	1547	1181	1633	1357	36.50	1302	1.131	1188	1.634	2.625
17		1.019	.164	478	.287	457	439	6554	536	1248	1127	1426	1308	34.86	1229	1.091	1129	1.578	2.586
18		1.021	.175	479	.289	458	438	5808	536	800	707	960	887	29.08	938	1.329	1024	1.562	2.333
19		1.022	.178	477	.284	457	440	5089	536	472	434	598	560	22.40	788	1.818	1011	1.212	2.298
20		1.022	.176	478	.285	457	440	3983	535	199	170	260	251	14.38	740	4.353	1065	1.066	2.489
21		1.021	.173	479	.286	457	440	3650	536	118	124	188	194	12.87	728	5.871	1102	1.058	2.505
22		1.025	.168	479	.287	457	440	5066	536	81	100	155	162	10.33	683	6.850	1130	1.053	2.568
23		1.290	.614	477	.350	410	448	7280	426	2277	2185	3185	3063	46.78	2615	1.206	1545	2.138	3.491
24		1.299	.625	475	.351	411	443	7280	426	2266	2159	3166	3059	46.75	2615	1.211	1540	2.138	3.478
25		1.309	.629	478	.380	410	442	7260	457	2012	1891	2933	2812	47.48	2220	1.174	1400	1.946	3.167
26		1.301	.625	479	.380	410	442	7260	465	1979	----	2894	----	47.46	2220	-----	1578	1.956	3.118
27		1.292	.616	482	.380	410	441	7280	510	1674	1538	2677	2441	47.81	1905	1.239	1282	1.742	2.862
28		1.299	.623	479	.380	411	445	7079	535	1451	1336	2338	2243	47.10	1713	1.282	1177	1.635	2.657
29		1.297	.621	478	.359	411	445	6718	536	1241	1137	2114	2012	45.58	1515	1.339	1103	1.556	2.490
30		1.291	.616	480	.358	410	441	6554	536	1143	1080	1988	1925	44.80	1427	1.321	1075	1.312	2.438
31		1.312	.638	475	.360	409	442	5808	535	----	533	----	1277	38.00	954	1.790	882	1.205	2.178
32		1.309	.633	476	.380	408	441	5089	535	----	171	----	765	29.98	702	4.106	881	1.015	1.852
33		1.296	.621	477	.350	412	444	3995	535	-39	-14	348	373	20.14	611	-----	747	.878	1.682
34		1.330	.652	477	.380	410	445	3086	535	-147	-188	171	150	15.81	670	-----	607	.799	1.364
35		1.866	.984	490	.482	390	465	7280	414	3028	2722	4958	4654	65.29	3520	1.293	1820	2.048	3.289
36		1.858	.984	478	.480	394	471	7280	414	2978	2705	4898	4625	64.56	3495	1.293	1520	2.057	3.227
37		1.852	.982	480	.478	398	478	7280	442	2587	2339	4514	4284	64.56	2975	1.261	1398	1.856	2.937
38		1.852	.982	478	.470	398	478	7280	494	2184	1938	4108	3860	64.47	2485	1.284	1256	1.852	2.844
39		1.843	.978	482	.475	395	471	7079	535	1886	1588	3783	3482	64.09	2120	1.588	1151	1.530	2.444
40		1.853	.982	479	.476	395	471	6718	535	1834	1218	3355	3059	61.24	1808	1.484	1060	1.411	2.261
41		1.848	.979	479	.475	395	471	6554	536	1519	1045	3075	2601	59.22	1640	1.569	1000	1.344	2.123
42		1.846	.976	478	.475	394	470	5808	535	483	278	1949	1742	48.51	915	3.316	800	1.018	1.702
43		1.868	.989	479	.481	396	479	5089	536	-35	-170	1185	1051	40.77	621	-----	580	.794	1.441
44	45,000	1.291	0.616	290	0.226	406	436	7260	436	1592	1274	1936	1818	28.80	1640	1.287	1504	2.135	3.450
45		1.288	.611	290	.224	407	437	7260	436	1401	1310	1940	1848	28.72	1649	1.259	1515	2.199	3.462
46		1.284	.609	292	.221	409	438	7260	467	1245	1141	1785	1679	28.70	1445	1.266	1395	1.945	3.178
47		1.268	.613	289	.210	408	439	7260	480	1183	1047	1721	1585	28.59	1374	1.312	1368	1.885	3.118
48		1.284	.619	289	.224	408	437	7260	538	1057	870	1602	1415	26.71	1218	1.394	1258	1.758	2.879
49		1.282	.607	291	.212	412	442	7079	538	992	820	1582	1360	26.24	1141	1.391	1218	1.699	2.781
50		1.285	.608	291	.212	411	441	6718	538	866	750	1572	1246	22.48	1000	1.370	1133	1.691	2.569
51		1.294	.621	289	.218	408	440	6554	538	760	661	1288	1169	22.61	926	1.404	1092	1.511	2.482
52		1.278	.606	291	.218	411	441	5808	535	400	513	818	731	22.45	675	1.315	938	1.323	2.127
53		1.302	.626	289	.221	407	439	5082	535	161	150	526	475	17.92	595	4.562	886	1.036	2.018
54	50,000	1.294	0.619	222	0.168	404	435	7260	442	1087	1002	1509	1424	22.88	1292	1.288	1532	2.096	3.522
55		1.301	.626	222	.168	405	436	7260	447	1081	987	1508	1414	22.53	1262	1.279	1522	2.088	3.499
56		1.289	.625	223	.168	405	437	7260	471	1087	1004	1494	1431	22.53	1232	1.227	1502	2.081	3.437
57		1.270	.595	227	.167	406	434	7260	483	953	860	1375	1270	22.49	1112	1.293	1386	1.911	3.194
58		1.294	.609	224	.168	406	436	7260	536	849	718	1256	1133	22.58	995	1.390	1290	1.755	2.968
59		1.288	.621	224	.171	405	433	7078	535	818	693	1251	1108	22.76	953	1.375	1234	1.712	2.850
60		1.280	.605	225	.150	446	479	6716	535	536	475	965	862	19.46	801	1.693	1214	1.498	2.334
61		1.300	.624	222	.152	440	474	6554	536	499	455	872	806	18.75	750	1.724	1147	1.407	2.420
62		1.283	.606	226	.147	451	484	5808	536	250	185	547	480	15.11	659	3.601	1027	1.189	2.122
63		1.295	.619	225	.147	451	486	5082	536	84	2	325	241	11.82	588	294.0	931	.962	1.916
64	55,000	1.300	0.635	168	0.125	410	443	7260	451	817	758	1145	1085	18.81	1058	1.411	1534	2.092	3.463
65		1.266	.591	168	.122	414	443	7260	455	818	757	1119	1058	18.46	1060	1.400	1544	2.134	3.485
66		1.324	.647	161	.122	406	442	7260	487	780	693	1112	1025	18.70	1002	1.446	1517	1.990	3.432
67		1.302	.626	163	.122	411	445	7260	527	668	581	988	901	18.57	908	1.563	1548	1.883	3.036
68		1.314	.637	162	.122	406	441	7079	535	580	556	964	879	18.53	854	1.559	1537	1.813	3.032
69		1.315	.638	164	.128	405	439	6718	535	523	417	858	732	18.12	779	1.868	1138	1.847	2.598
70		1.304	.628	169	.130	406	438	6554	535	482	385	788	688	15.49	744	1.932	1126	1.570	2.571
71		1.296	.619	168	.129	407	438	5808	535	264	240	507	485	12.85	655	2.729	977	1.275	2.231
72		1.279	.604	170	.124	410	440	5082	535	124	81	309	276	9.98	637	7.000	924	1.071	2.100

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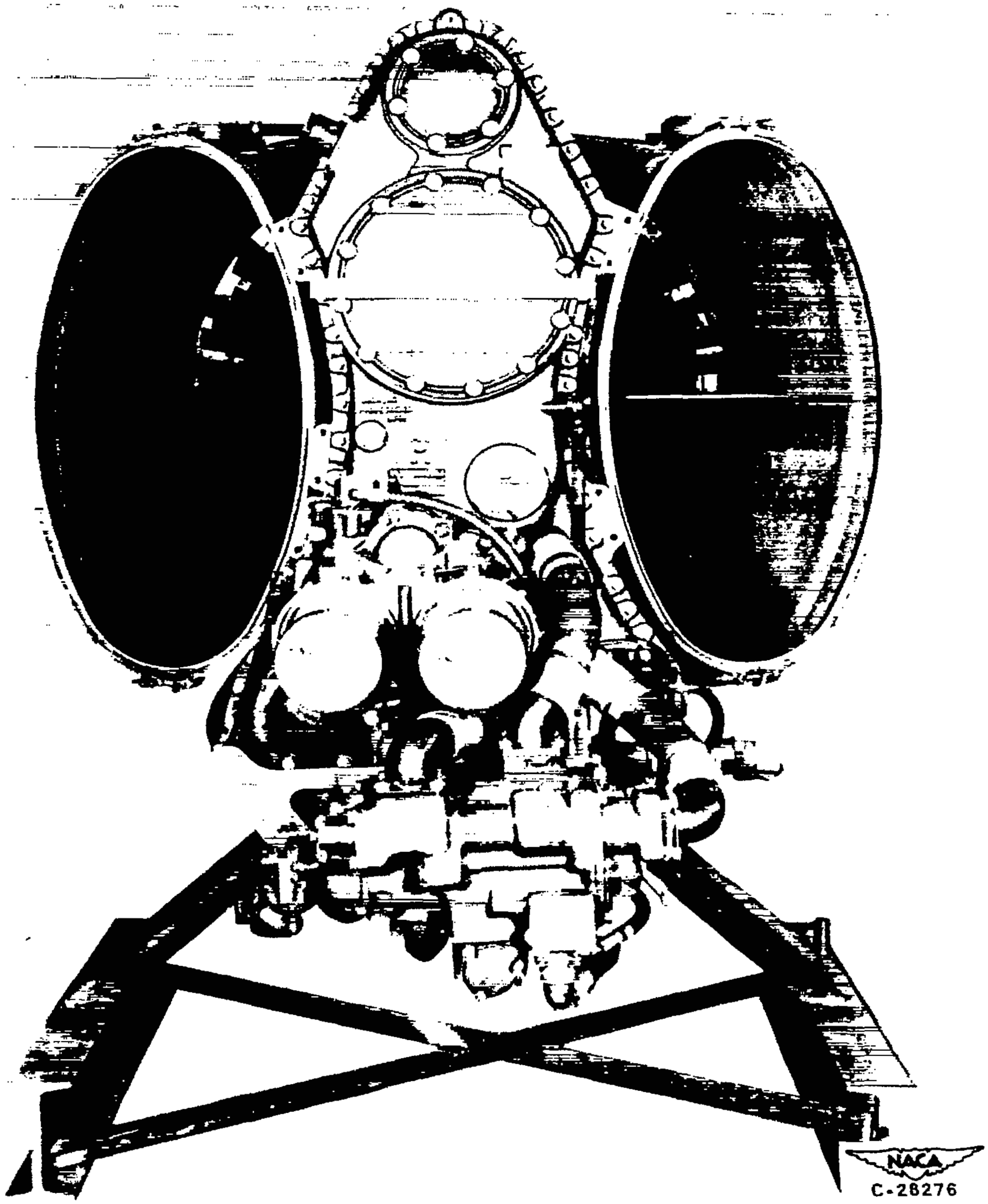

Figure 1. - View looking downstream of inlet of prototype J40-WE-8 turbojet engine.

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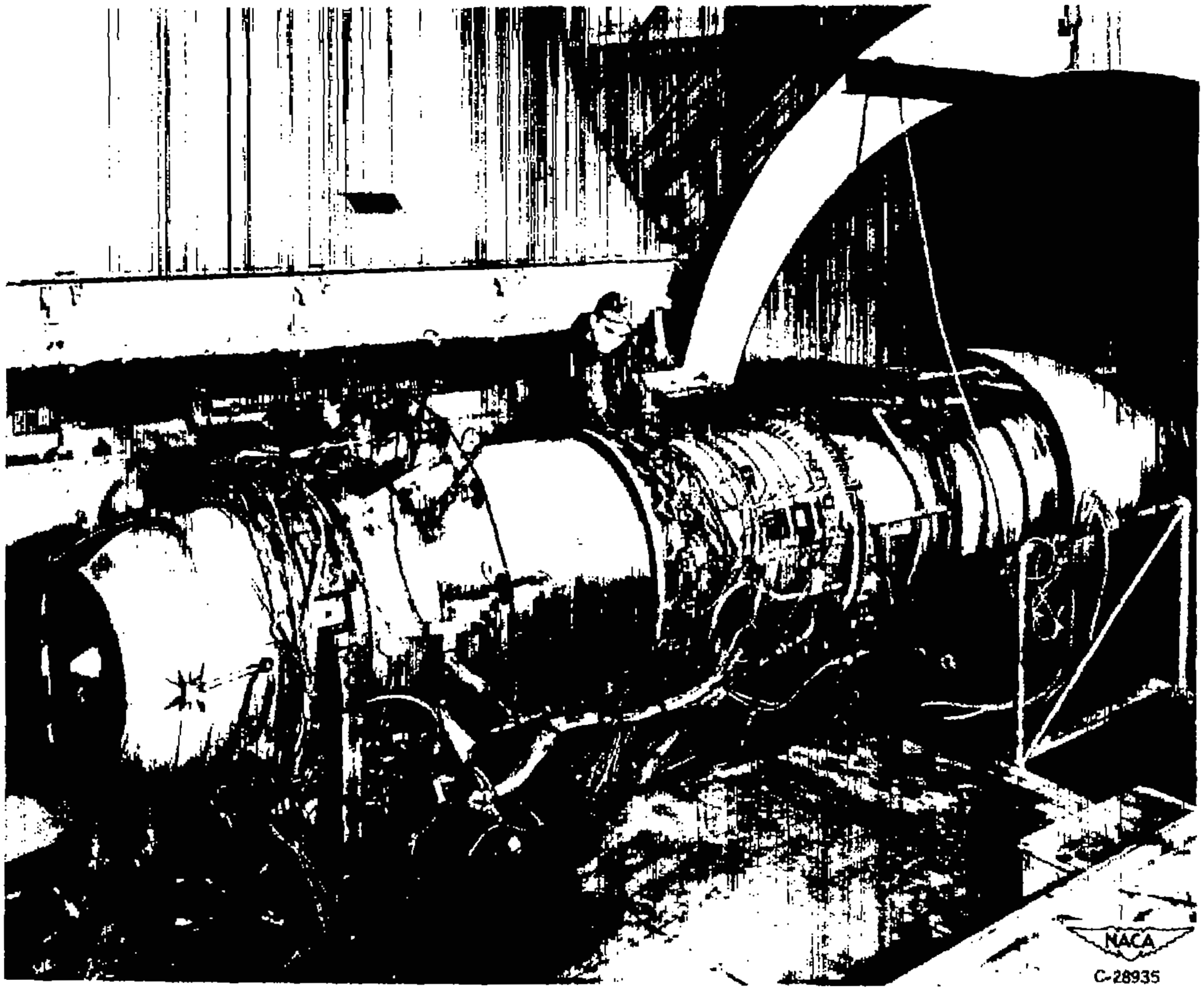

Figure 2. - Prototype J40-WE-8 turbojet engine installed in test section of altitude wind tunnel.

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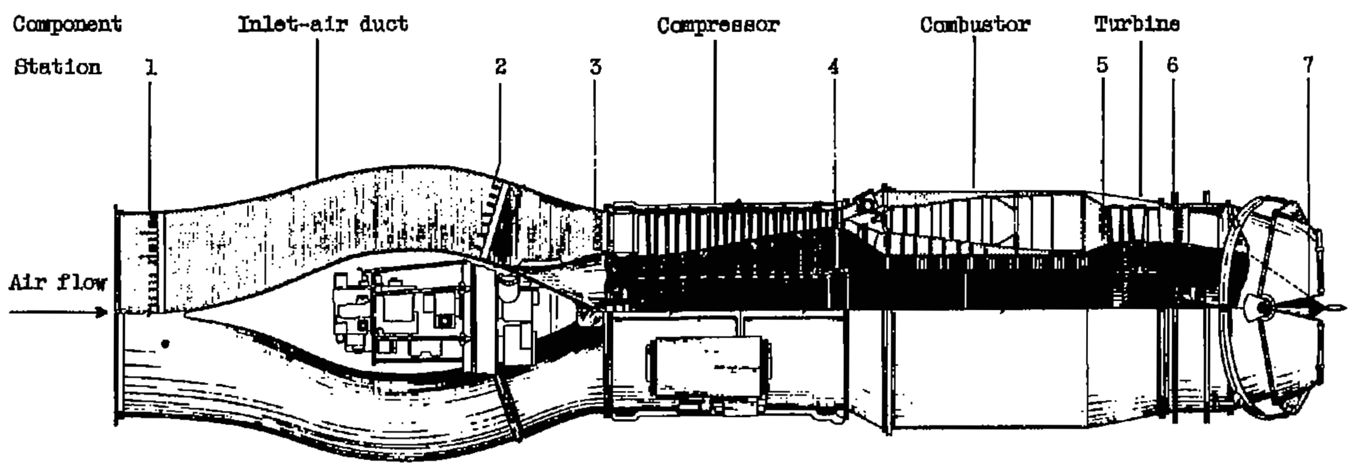

Station	Location	Total-pressure tubes	Static-pressure tubes	Wall static-pressure orifices	Thermo-couples
1	Inlet-air duct	29	12	6	10
2	Engine inlet	18	0	4	0
3	Compressor inlet	23	3	7	0
4	Compressor outlet	18	0	3	6
5	Turbine inlet	5	0	0	10[a]
6	Turbine outlet	20	0	8	24
7	Exhaust-nozzle inlet	16	2	8	0

[a] Sonic flow probes

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Figure 3. - Top view of prototype J40-WE-8 turbojet-engine installation showing stations at which instrumentation was installed.

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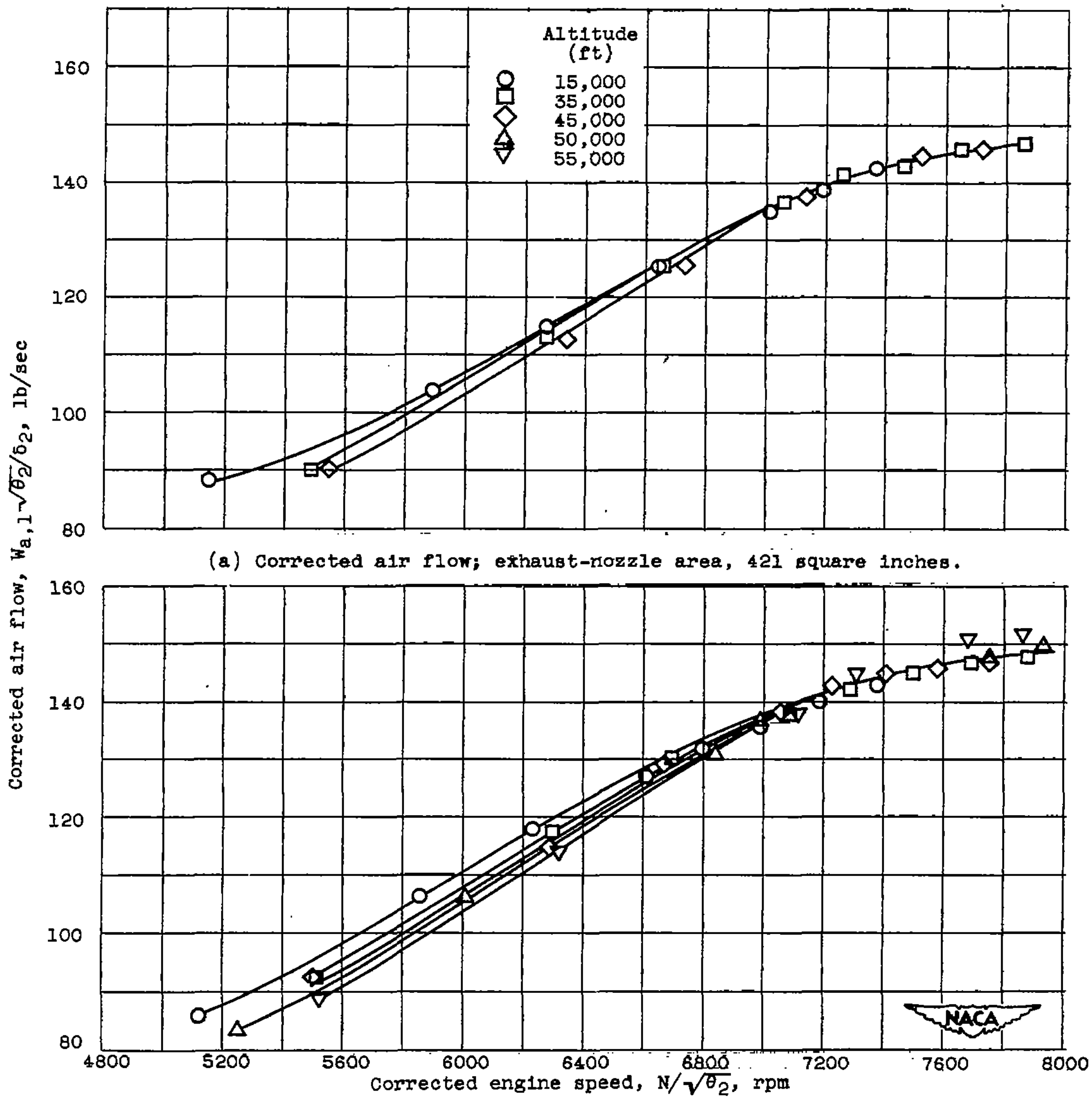

(a) Corrected air flow; exhaust-nozzle area, 421 square inches.

(b) Corrected air flow; exhaust-nozzle area, 535 square inches.

Figure 4. - Effect of altitude on corrected engine performance at flight Mach number of 0.62.

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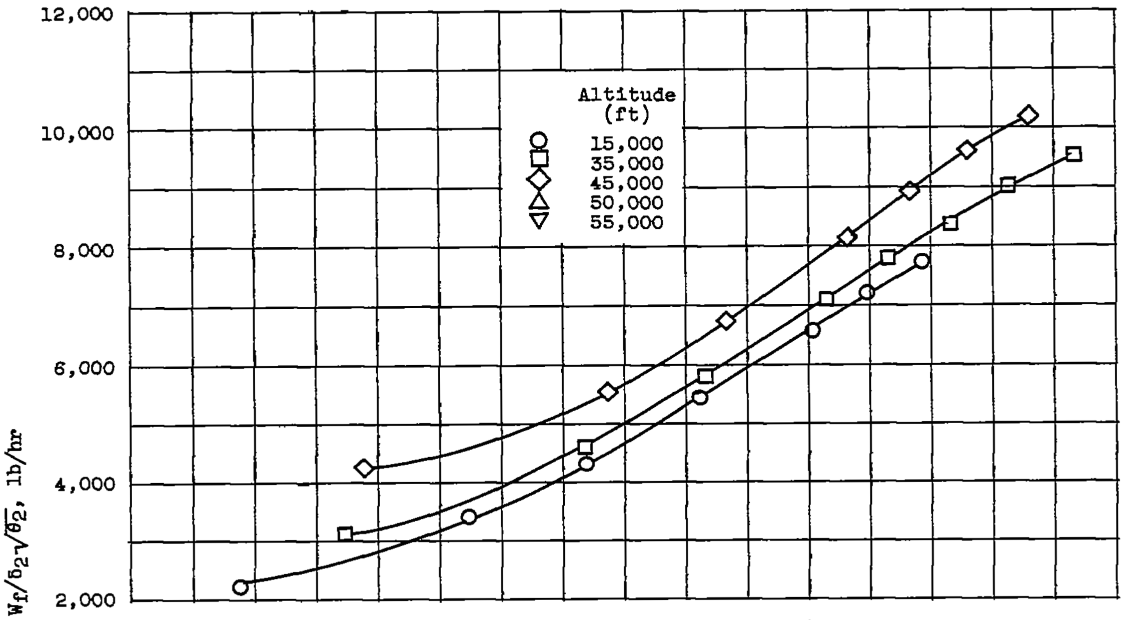
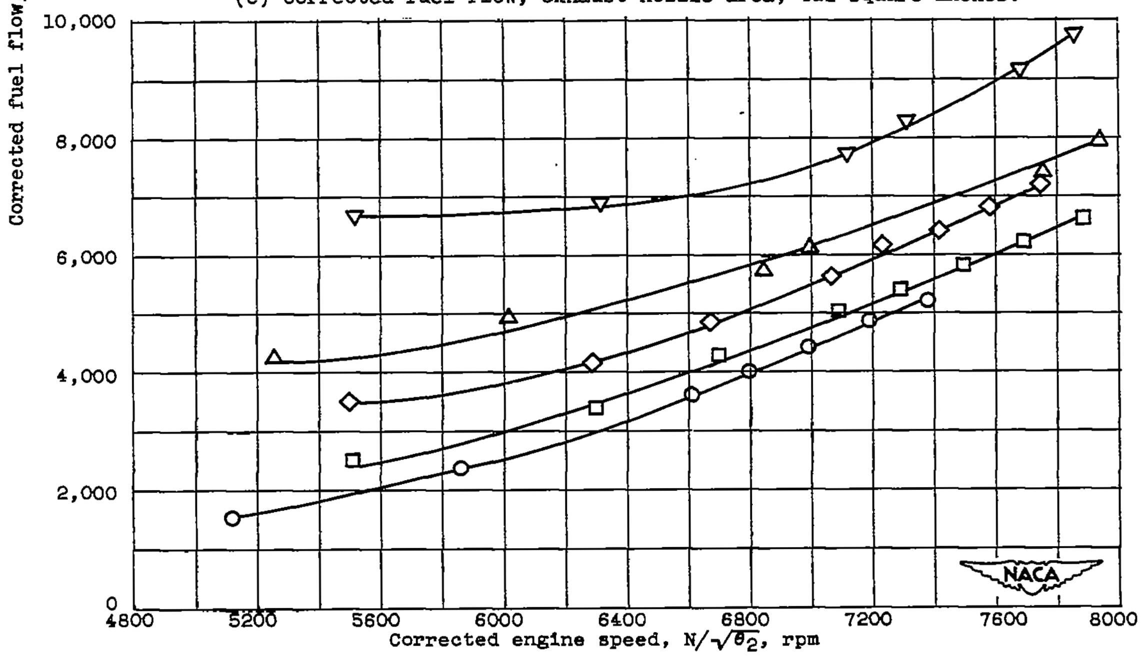
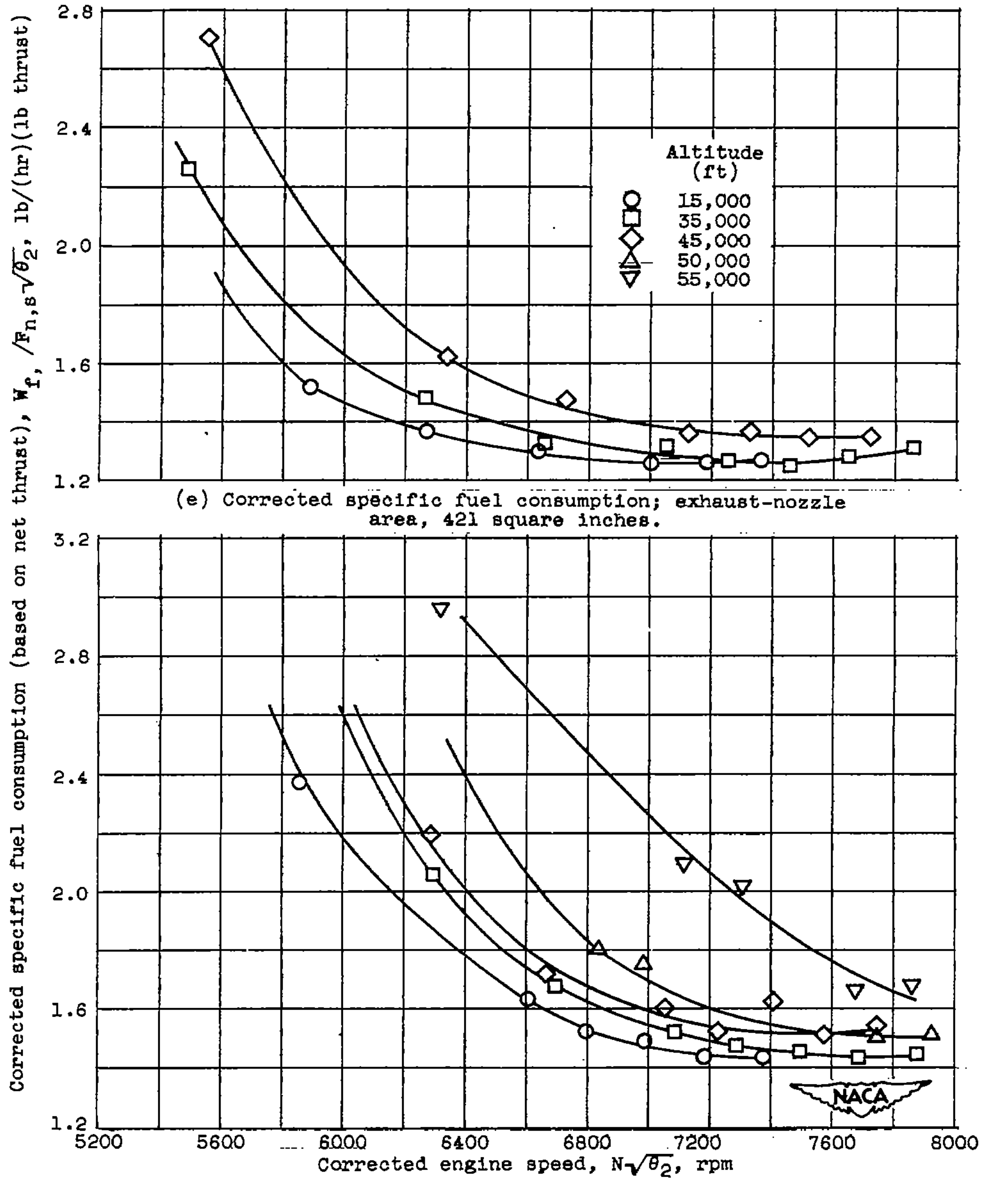


(c) Corrected fuel flow; exhaust-nozzle area, 421 square inches.



(d) Corrected fuel flow; exhaust-nozzle area, 535 square inches.

Figure 4. - Continued. Effect of altitude on corrected engine performance at flight Mach number of 0.62.



(e) Corrected specific fuel consumption; exhaust-nozzle area, 421 square inches.

(f) Corrected specific fuel consumption; exhaust-nozzle area, 535 square inches.

Figure 4. - Continued. Effect of altitude on corrected engine performance at flight Mach number of 0.62.

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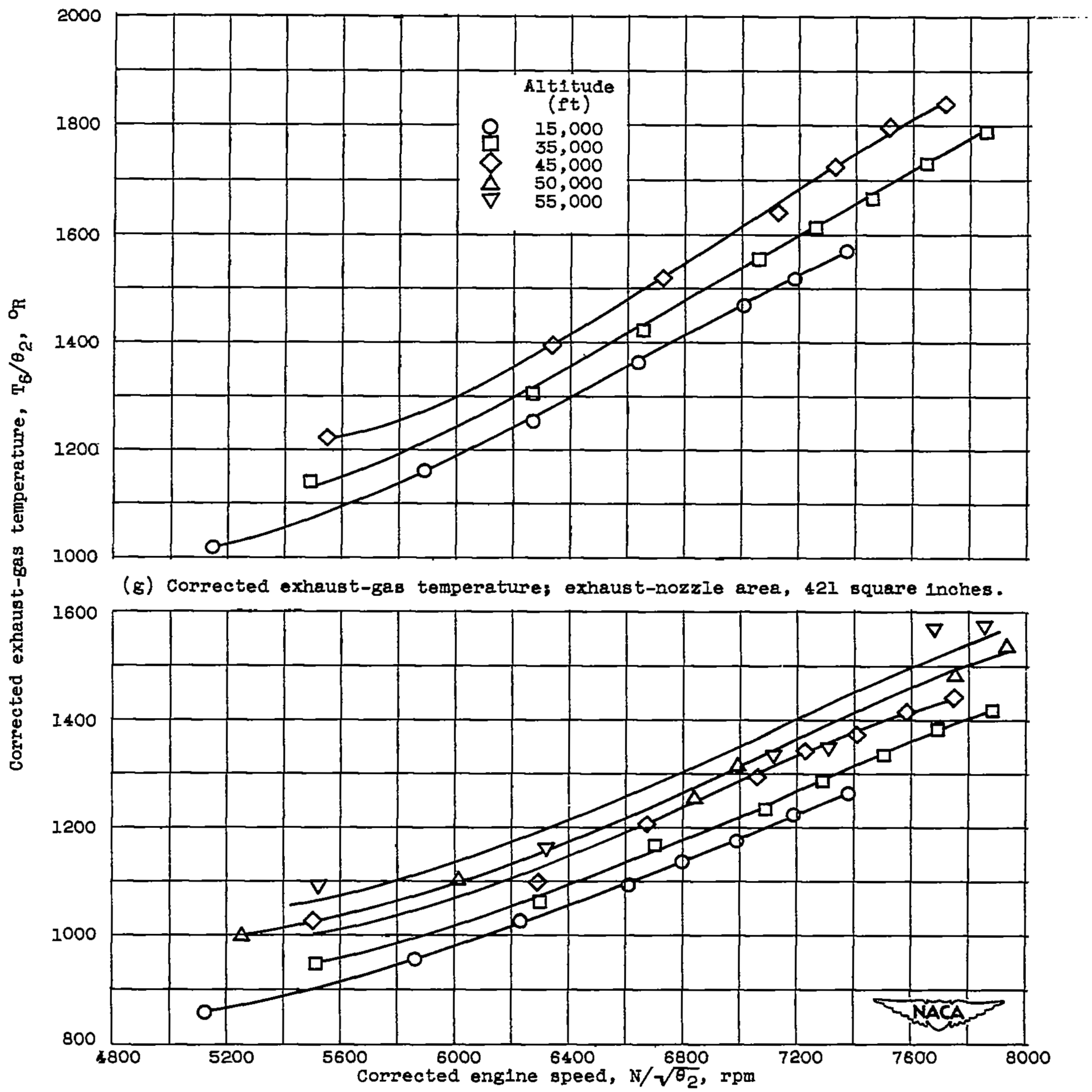
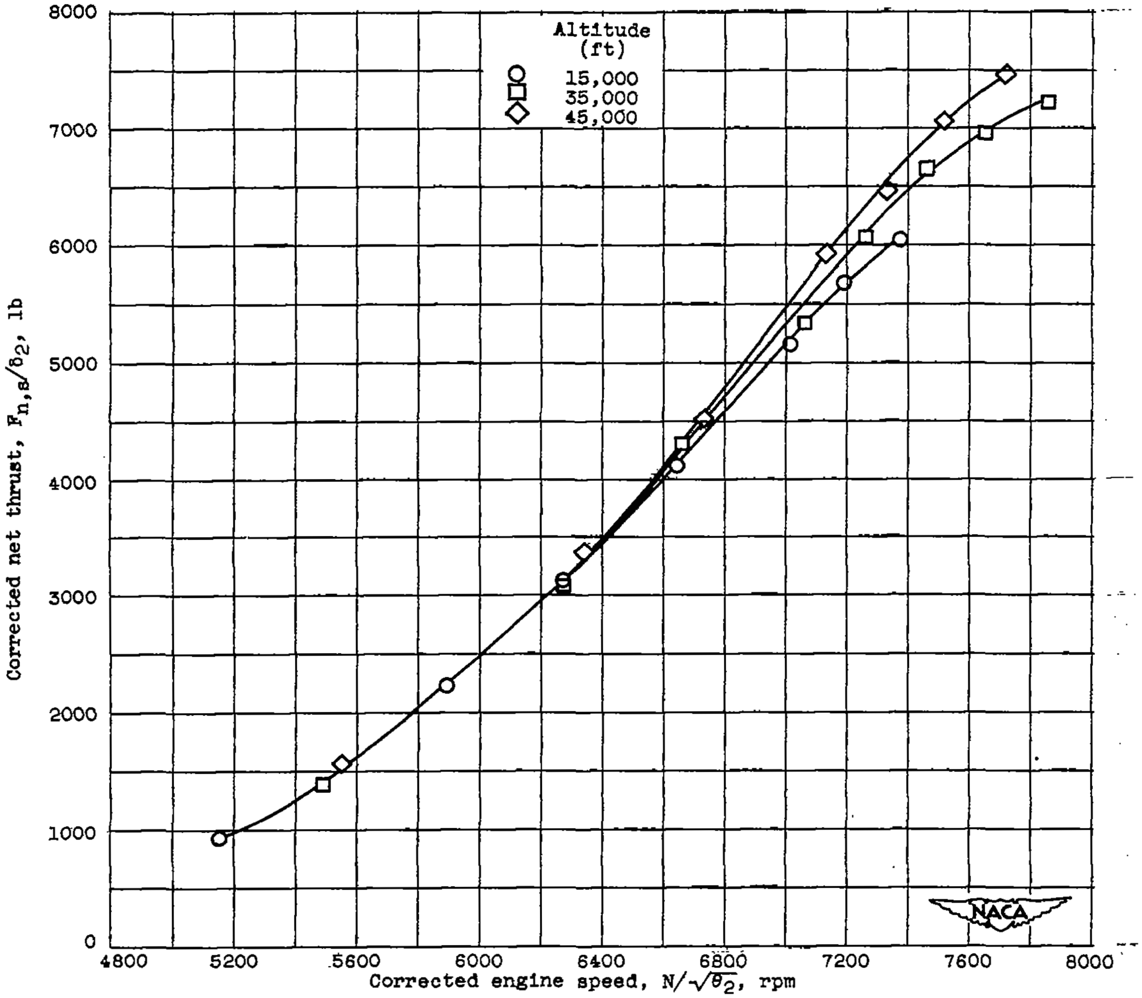


(g) Corrected exhaust-gas temperature; exhaust-nozzle area, 421 square inches.

(h) Corrected exhaust-gas temperature; exhaust-nozzle area, 535 square inches.

Figure 4. - Continued. Effect of altitude on corrected engine performance at flight Mach number of 0.62.



(i) Corrected net thrust; exhaust-nozzle area, 421 square inches.

Figure 4. - Continued. Effect of altitude on corrected engine performance at a flight Mach number of 0.62.

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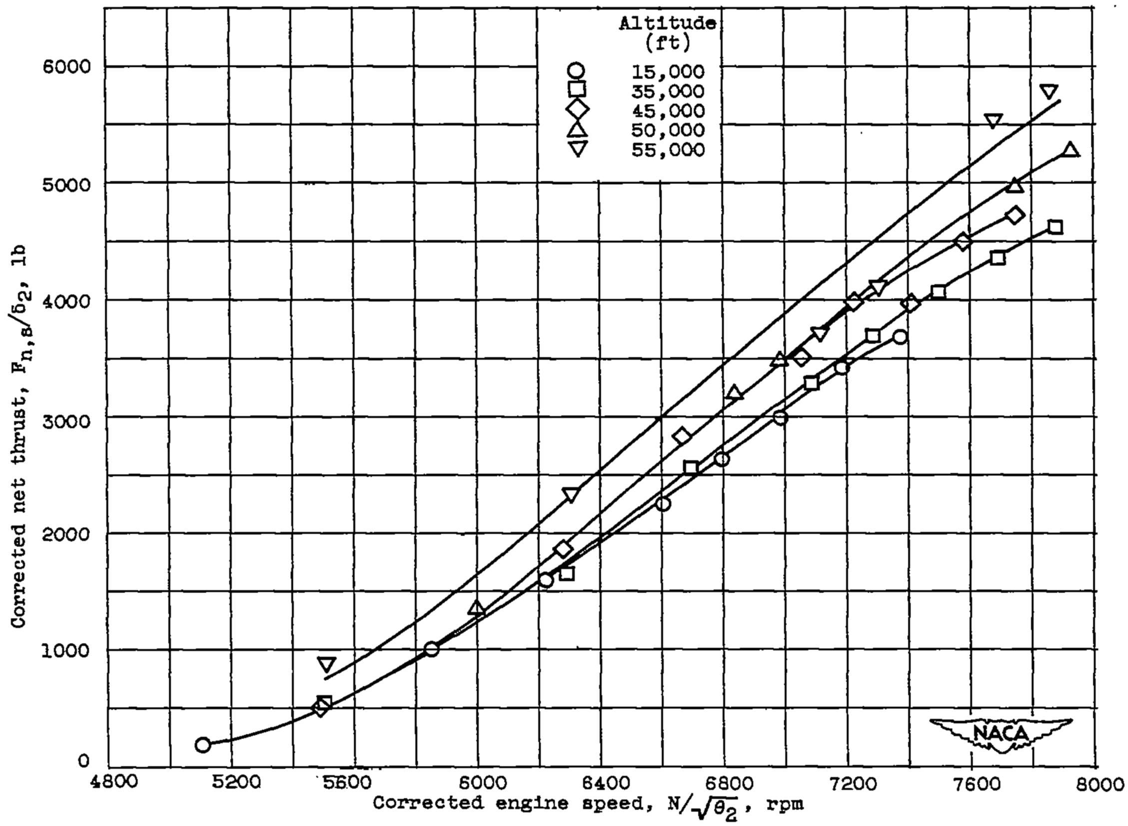
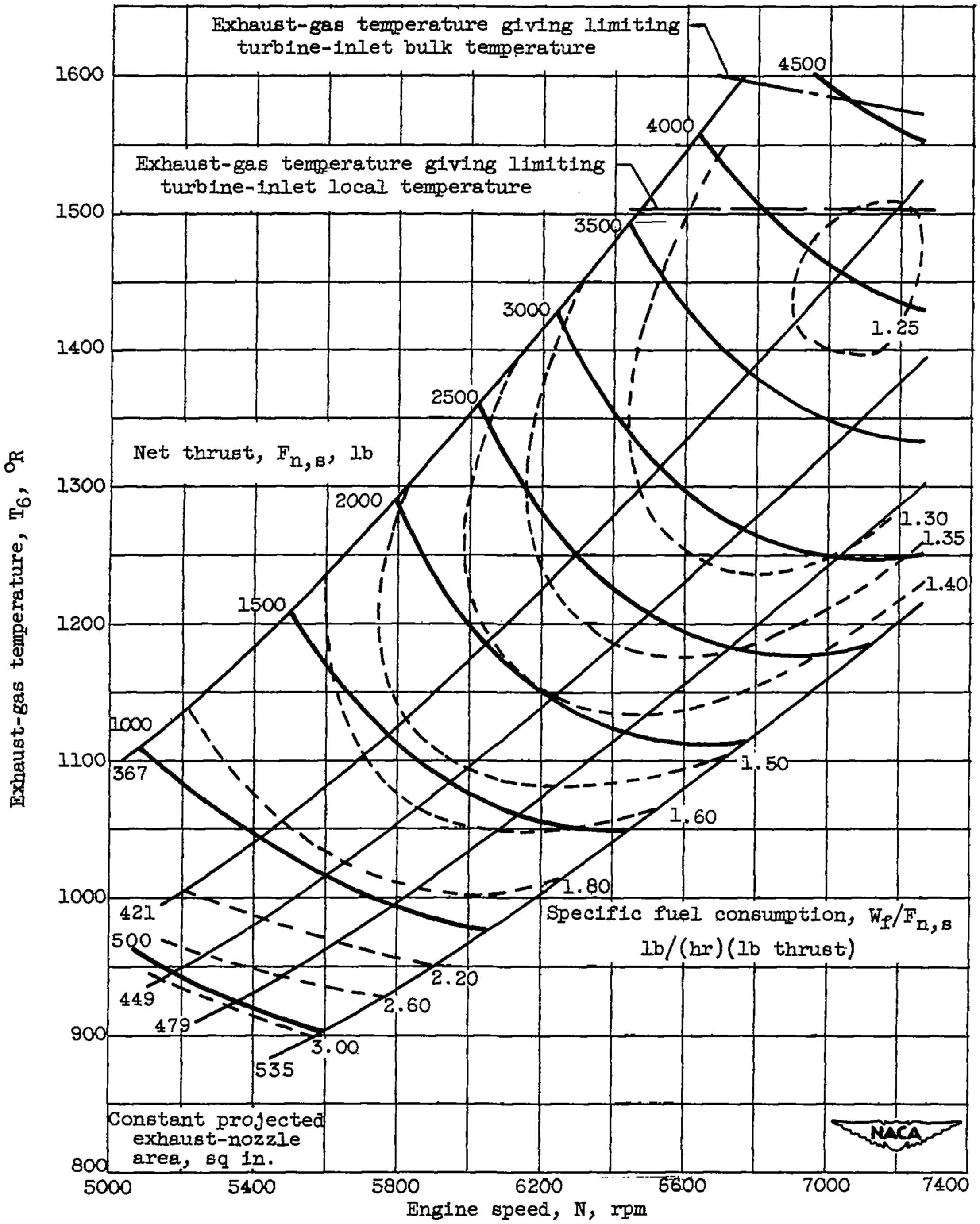


(j) Corrected net thrust; exhaust-nozzle area, 535 square inches.

Figure 4. - Concluded. Effect of altitude on corrected engine performance at a flight Mach number of 0.62.



(a) Altitude, 15,000 feet; flight Mach number, 0.62; equivalent inlet-air temperature, 468° R.

Figure 5. - Engine performance maps.

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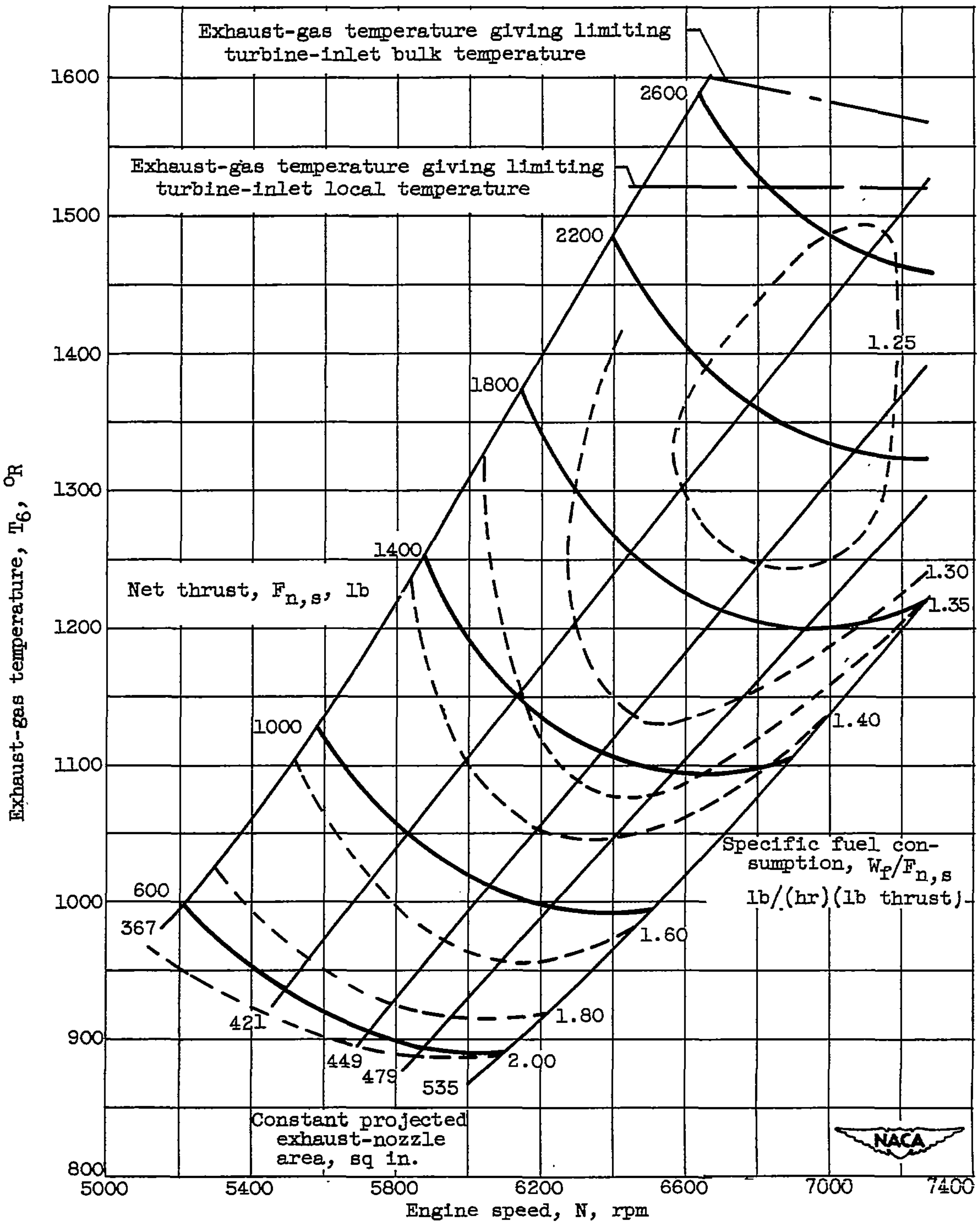

(b) Altitude, 35,000 feet; flight Mach number, 0.99; equivalent inlet-air temperature, 393° R.

Figure 5. - Continued. Engine performance maps.

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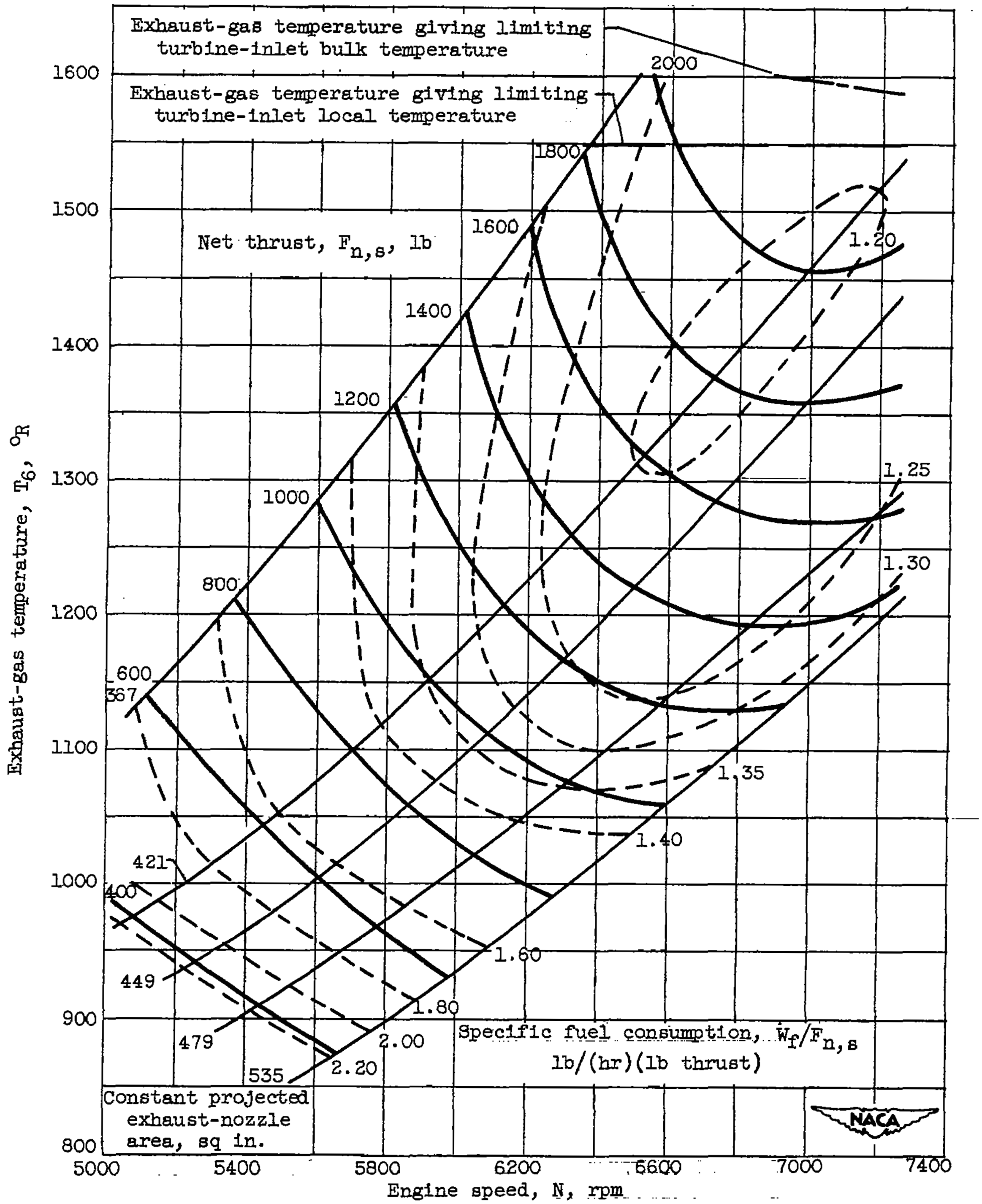

(c) Altitude, 35,000 feet; flight Mach number, 0.62; equivalent inlet-air temperature, 414° R.

Figure 5. - Continued. Engine performance maps.

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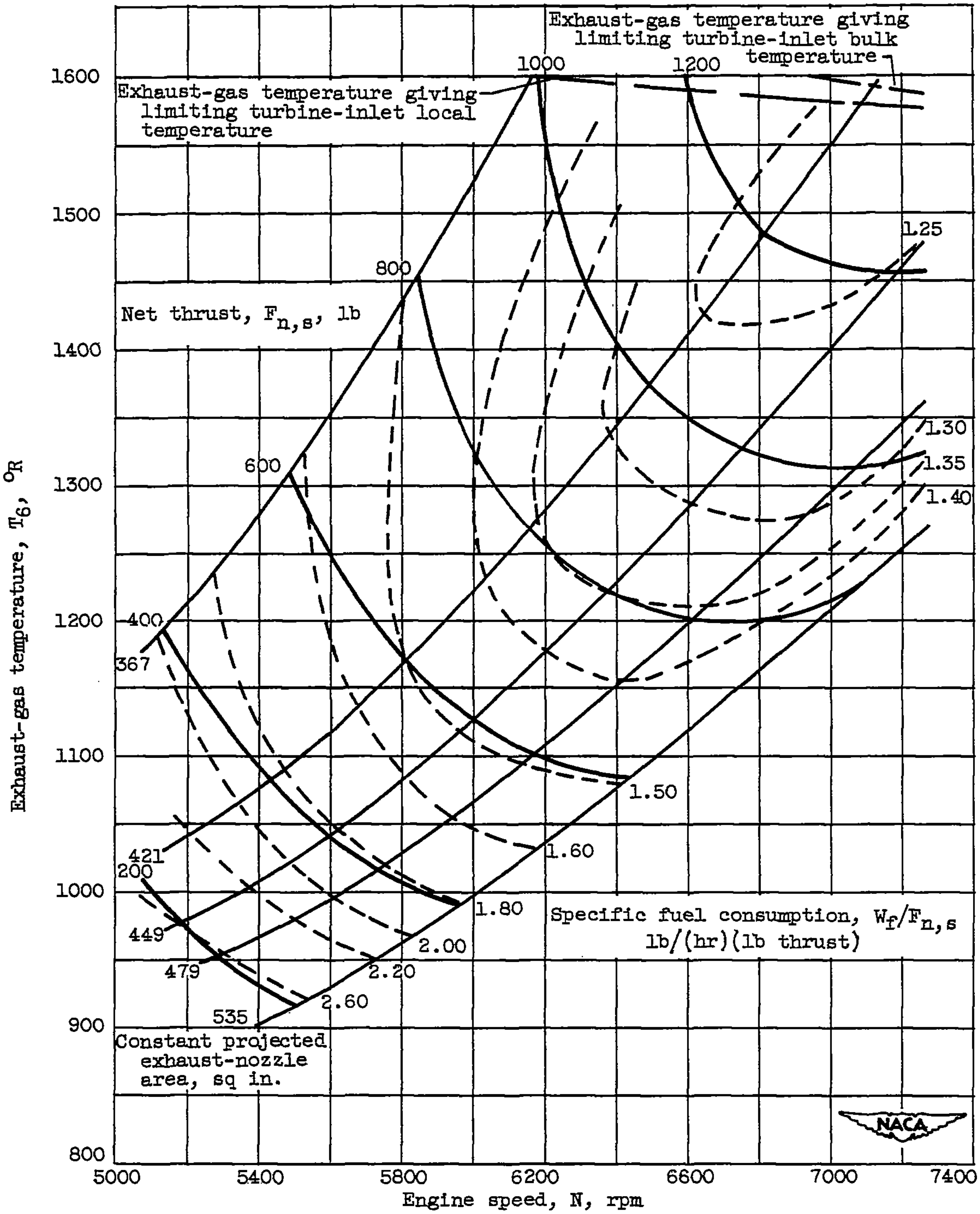
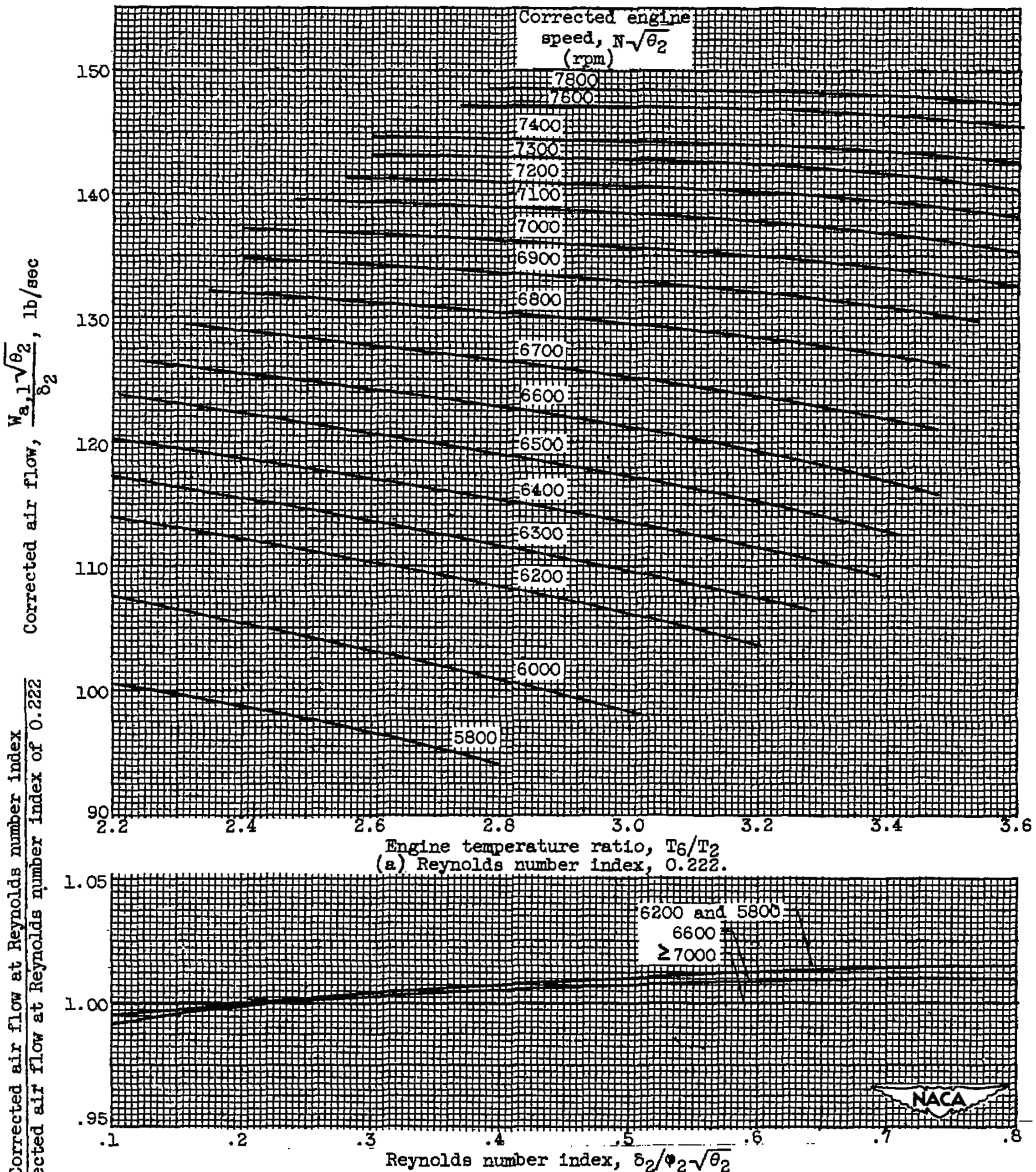


(d) Altitude, 45,000 feet; flight Mach number, 0.62; equivalent inlet-air temperature, 410° R.

Figure 5. - Concluded. Engine performance maps.

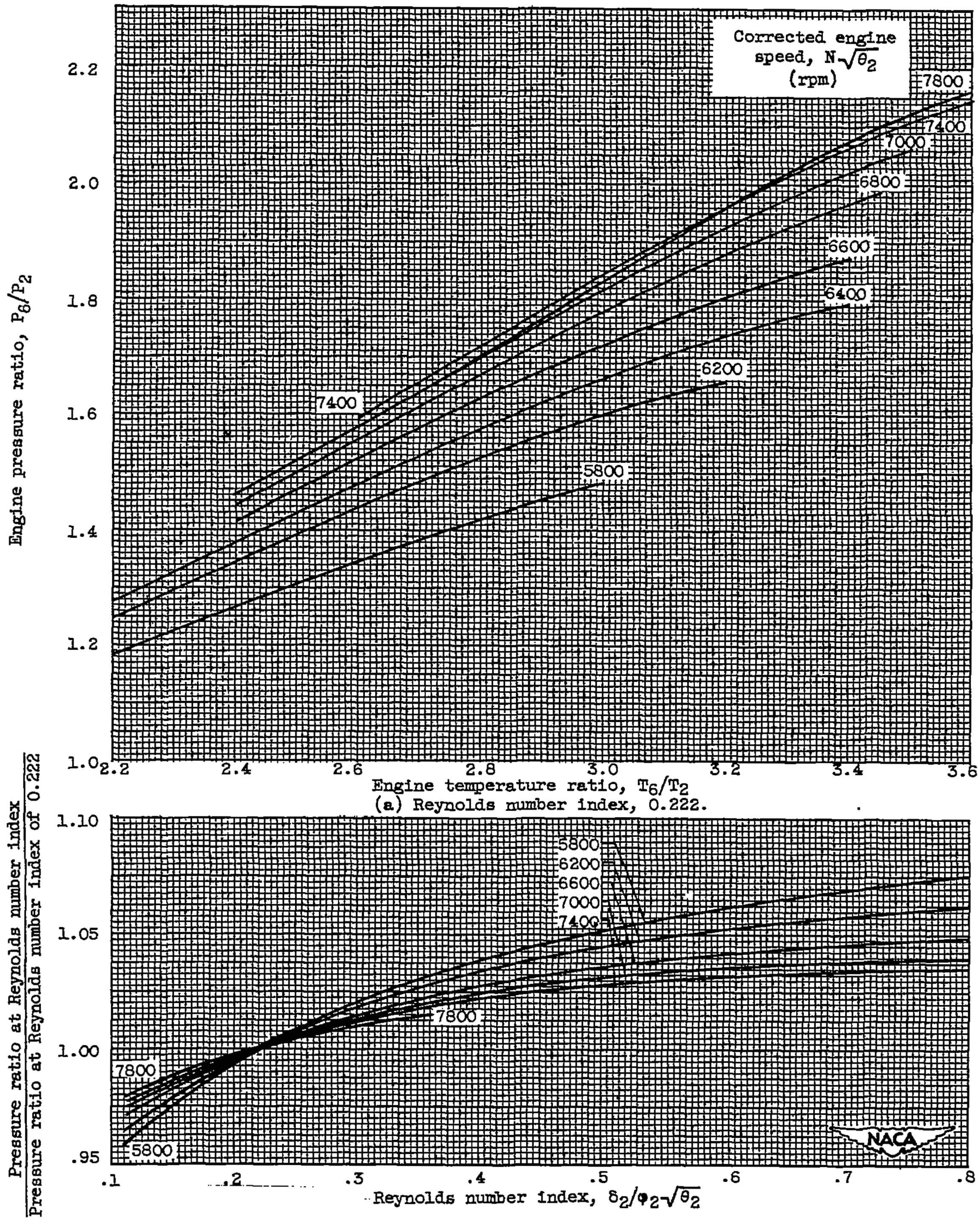


(a) Reynolds number index, 0.222.

(b) Correction factor for Reynolds number index.

Figure 6. - Variation of corrected air flow with Reynolds number index, corrected engine speed, and engine temperature ratio.

2733



(a) Reynolds number index, 0.222.

(b) Correction factor for Reynolds number index.

Figure 7. - Variation of engine pressure ratio with Reynolds number index, corrected engine speed, and engine temperature ratio.

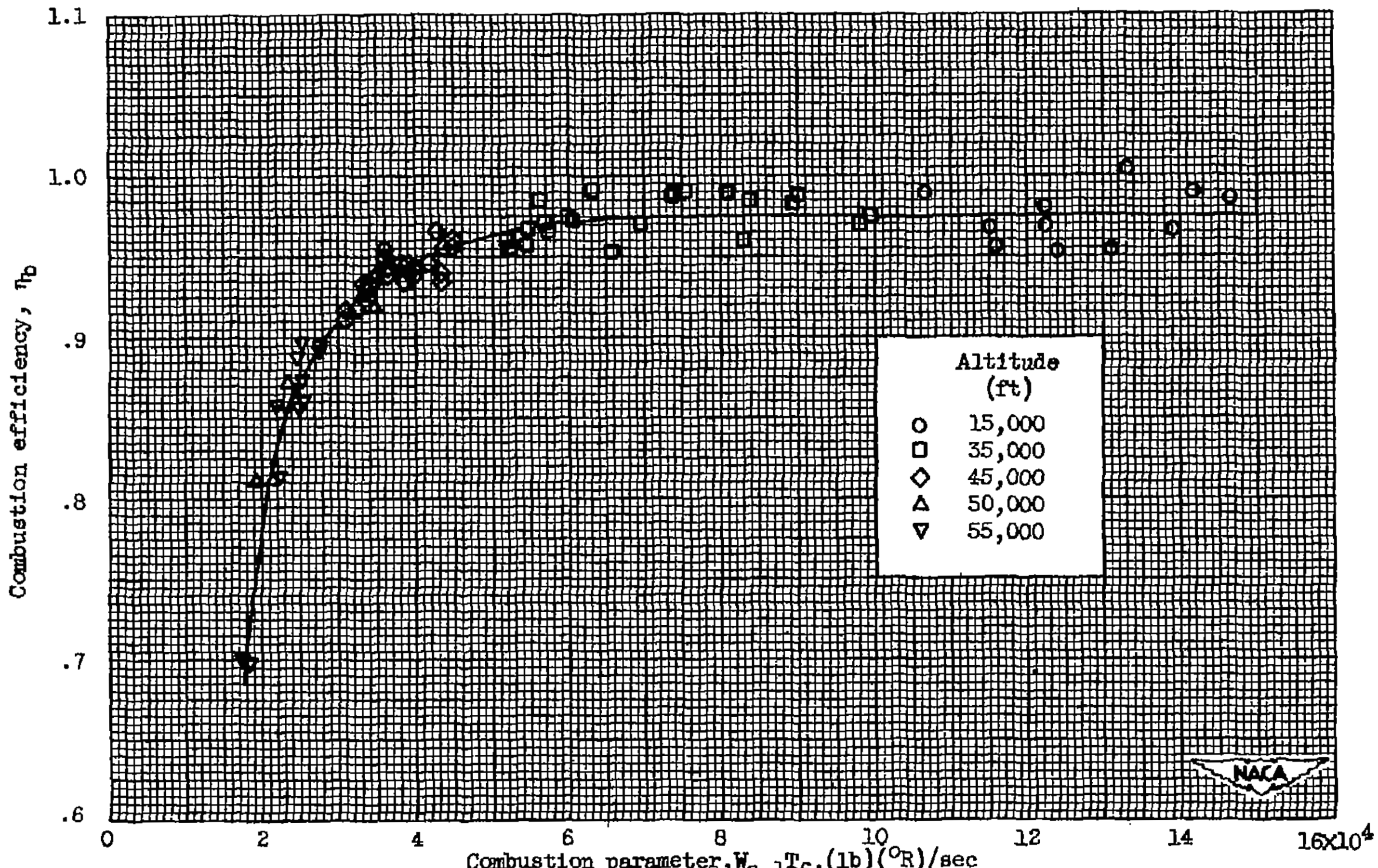


Figure 8. - Variation of combustion efficiency with combustion parameter.

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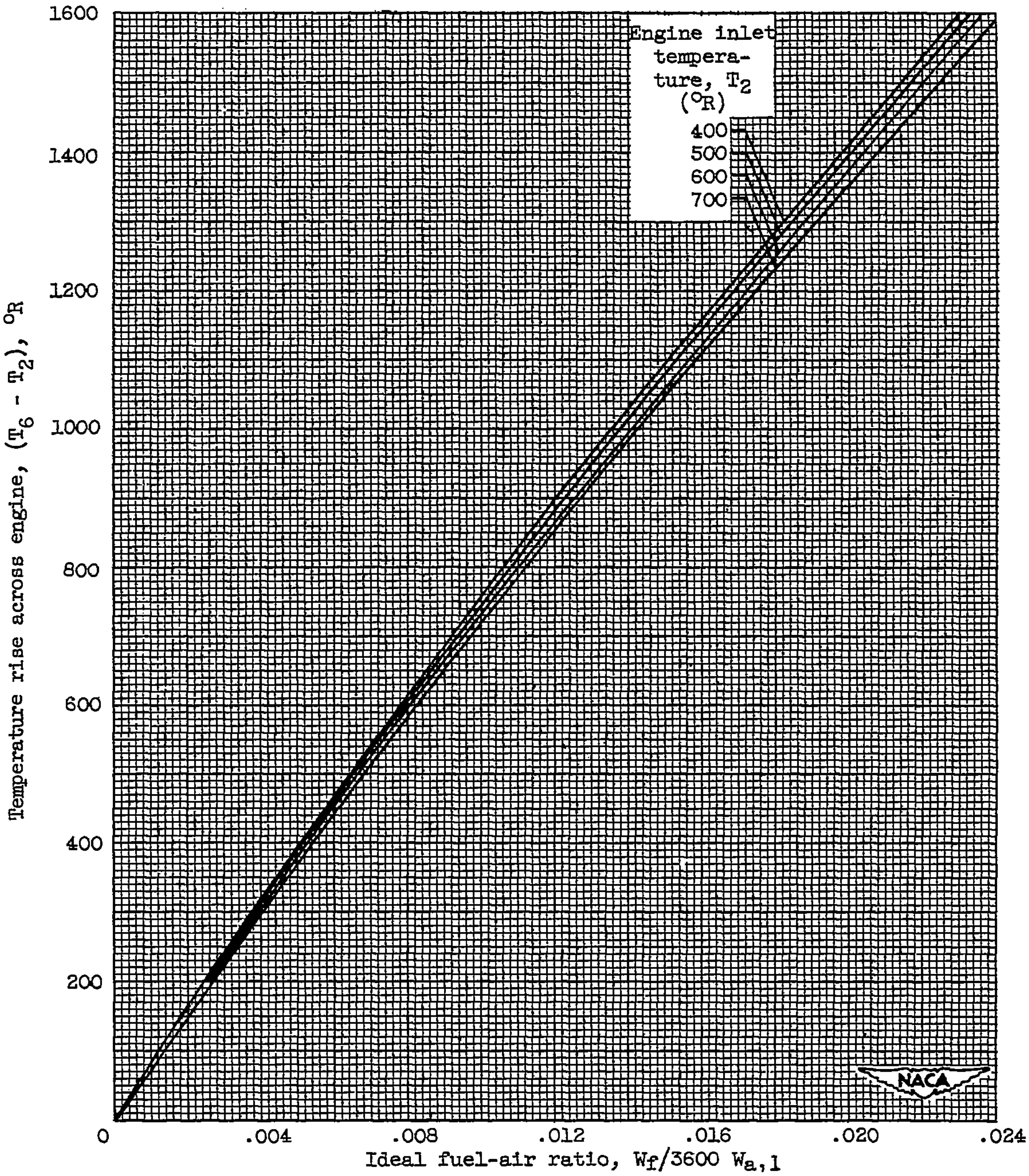

Figure 9. - Engine temperature rise as function of fuel-air ratio. Lower heating value, 18,700 Btu per pound; hydrogen-carbon ratio, 0.171.

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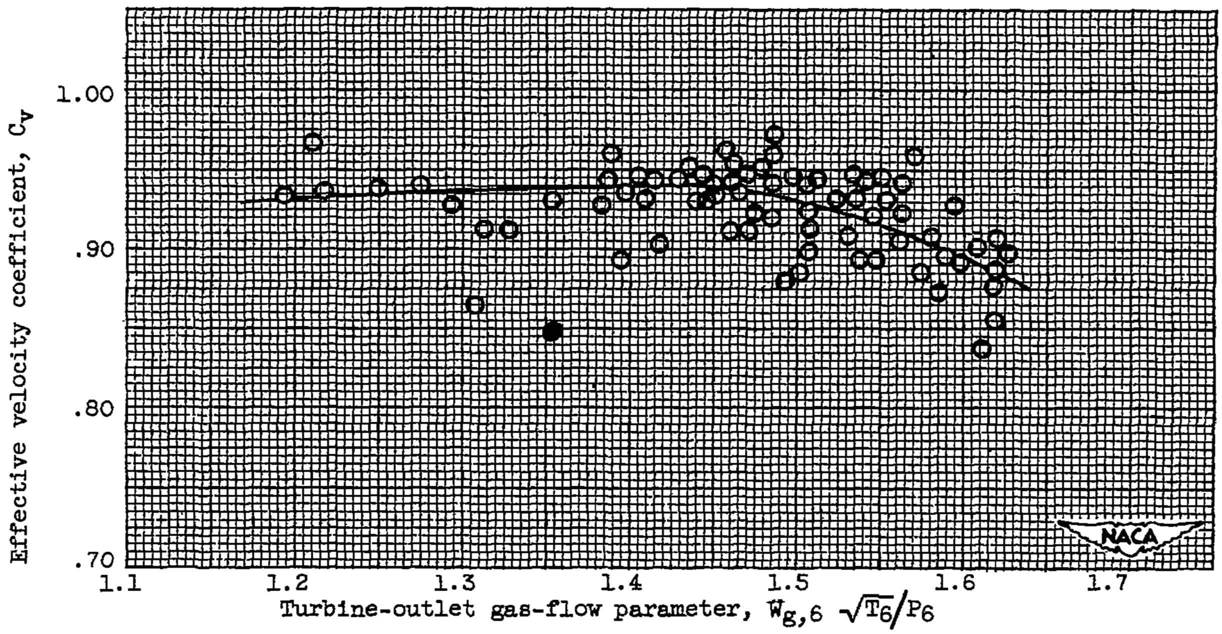

Figure 10. - Variation of effective velocity coefficient with turbine-outlet gas-flow parameter.

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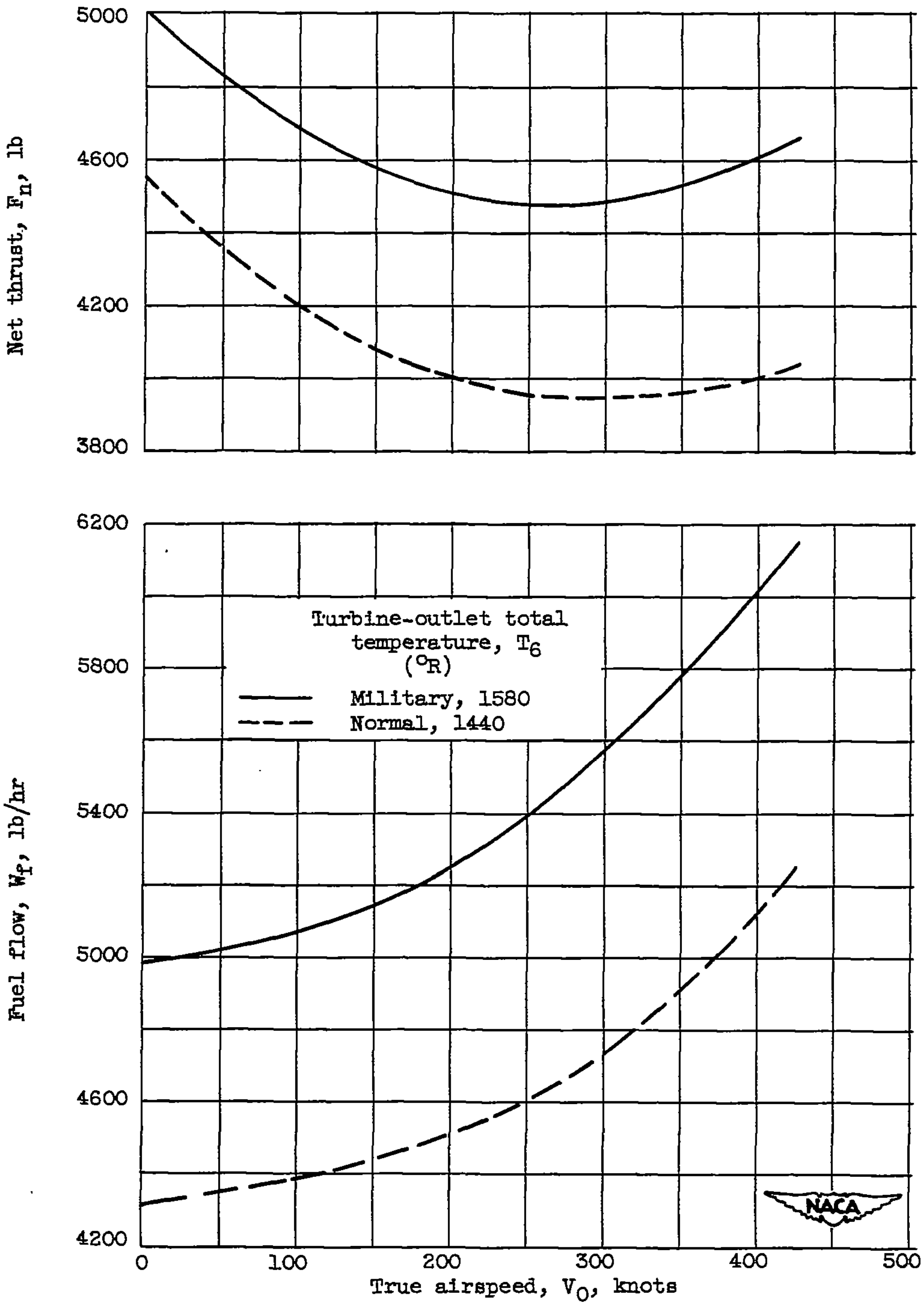
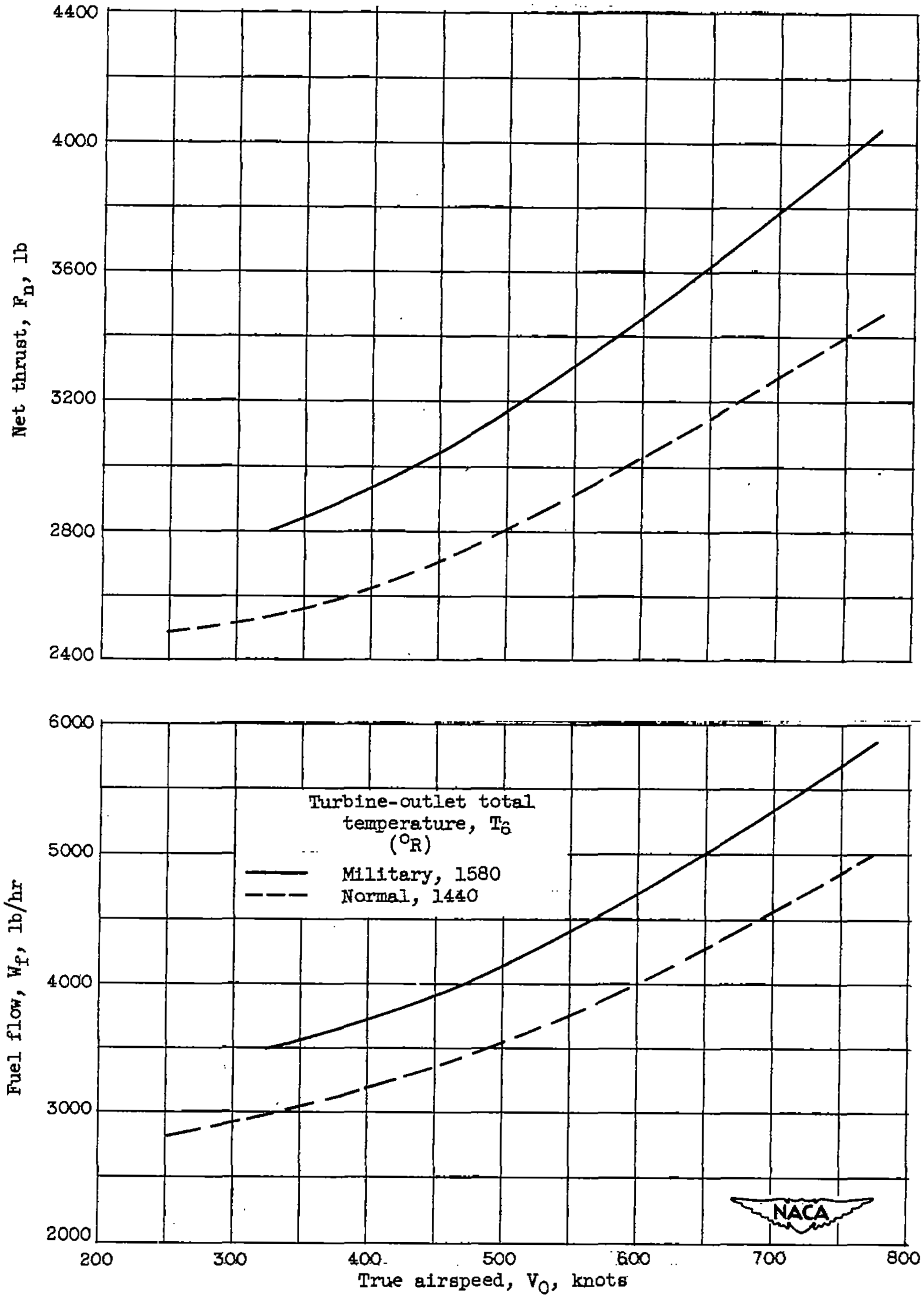


(a) Altitude, 15,000 feet.

Figure 11. - Variation of net thrust and fuel flow with flight speed obtained by calculation from pumping characteristics. Engine speed, 7260 rpm.

2733



(b) Altitude, 30,000 feet.

Figure 11. - Continued. Variation of net thrust and fuel flow with flight speed obtained by calculation from pumping characteristics. Engine speed, 7260 rpm.

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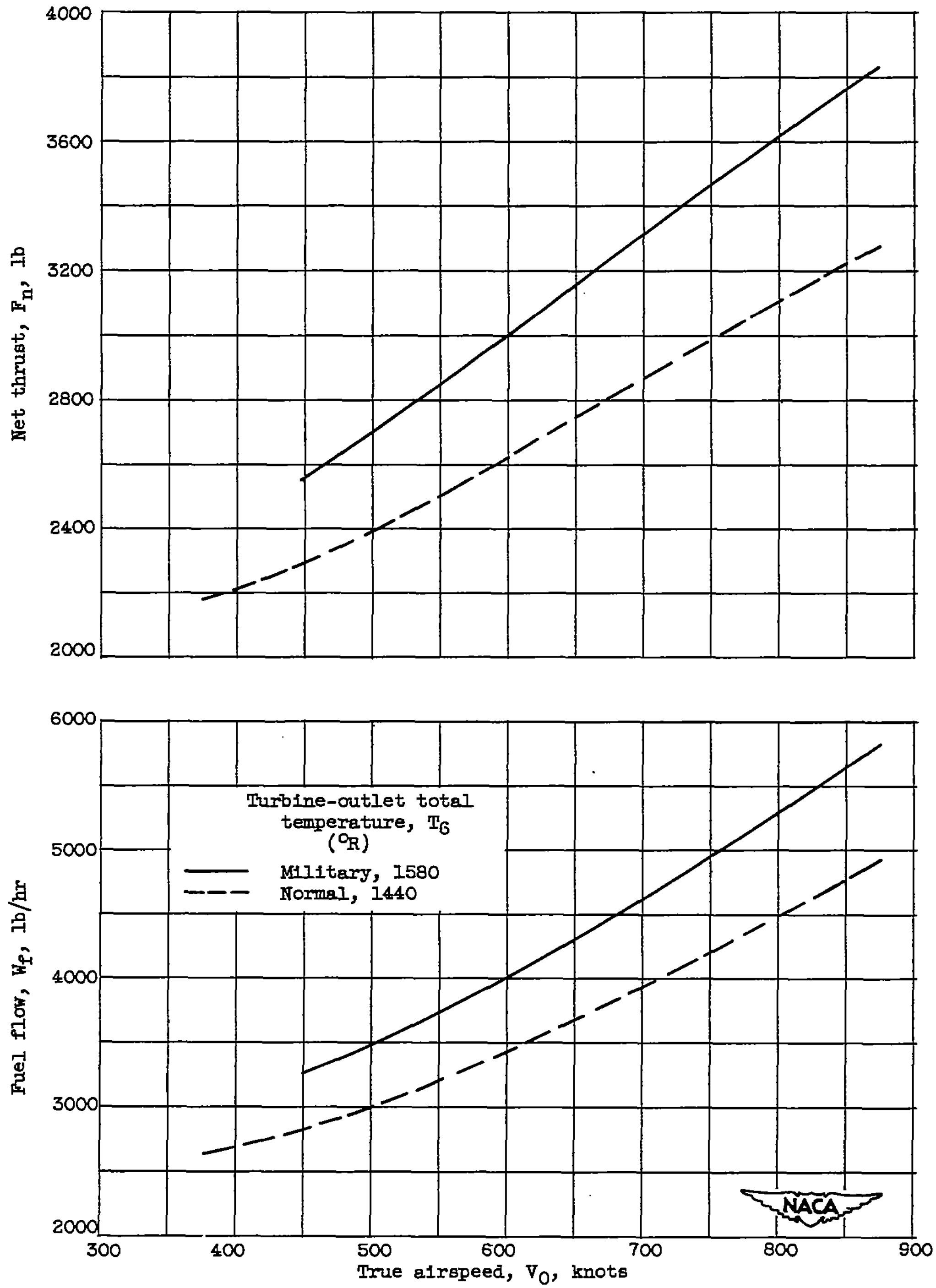
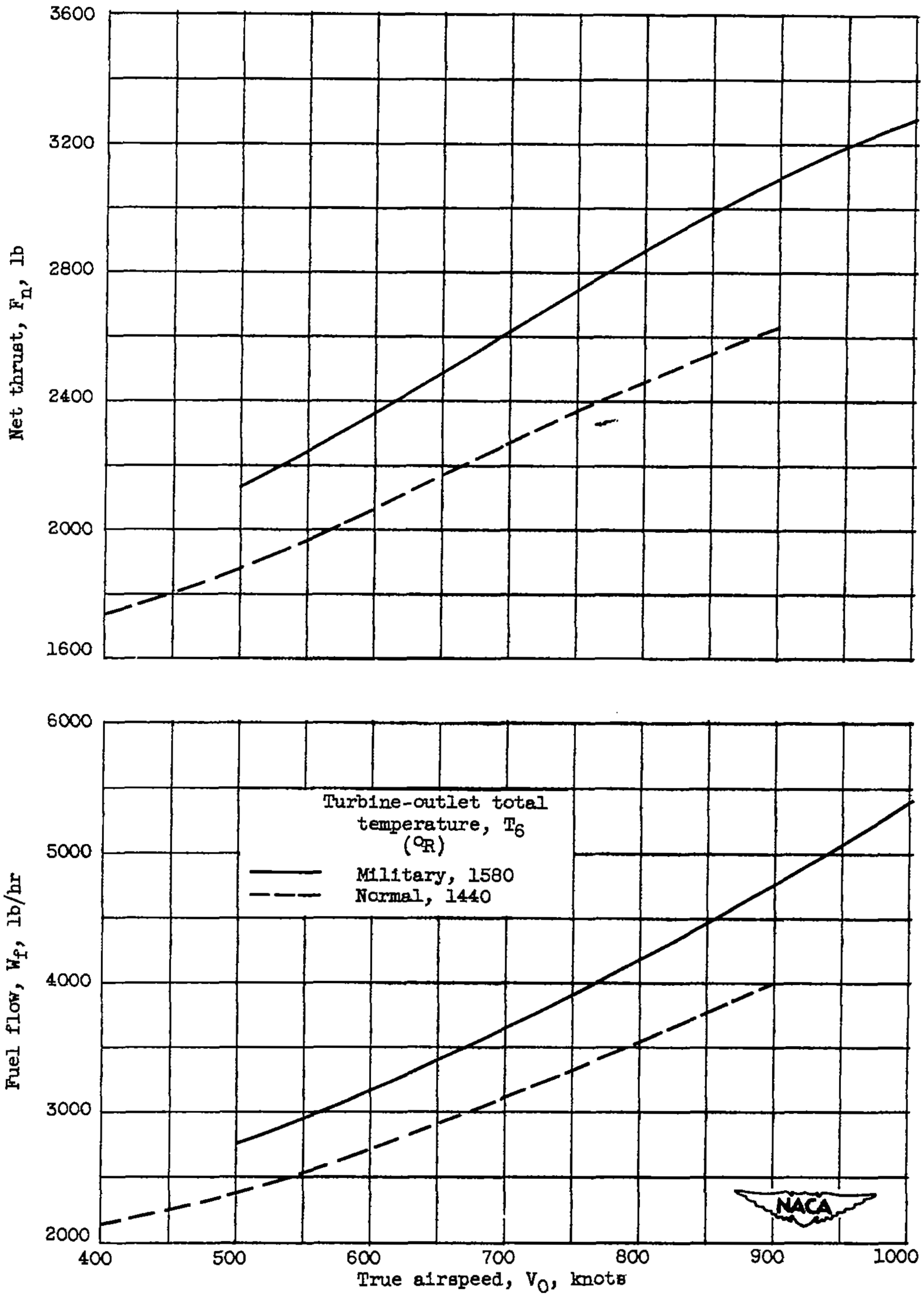


(c) Altitude, 35,000 feet.

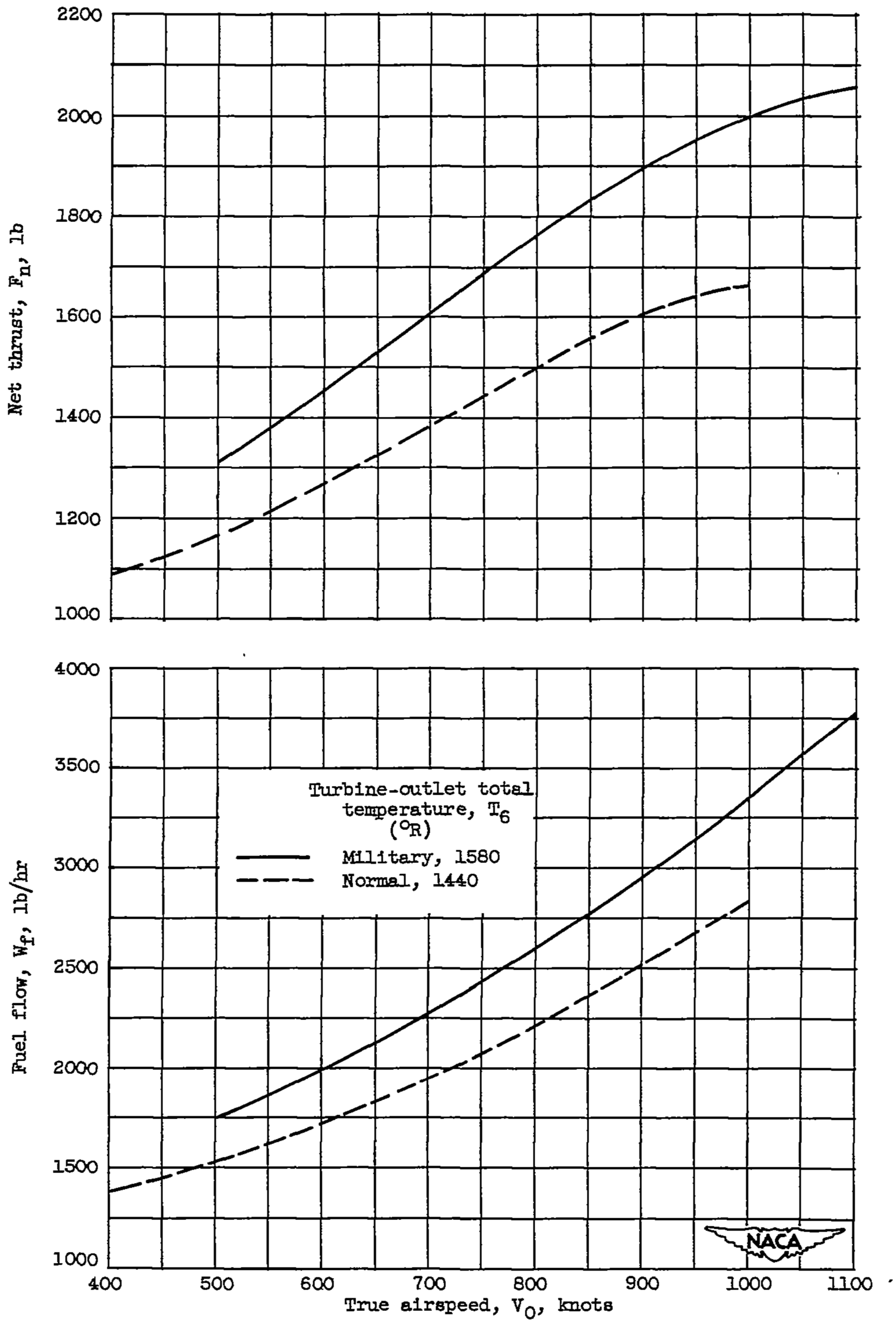
Figure 11. - Continued. Variation of net thrust and fuel flow with flight speed obtained by calculation from pumping characteristics. Engine speed, 7260 rpm.

2733



(d) Altitude, 40,000 feet.

Figure 11. - Continued. Variation of net thrust and fuel flow with flight speed obtained by calculation from pumping characteristics. Engine speed, 7260 rpm.



(e) Altitude, 50,000 feet.

Figure 11. - Concluded. Variation of net thrust and fuel flow with flight speed obtained by calculation from pumping characteristics. Engine speed, 7260 rpm.

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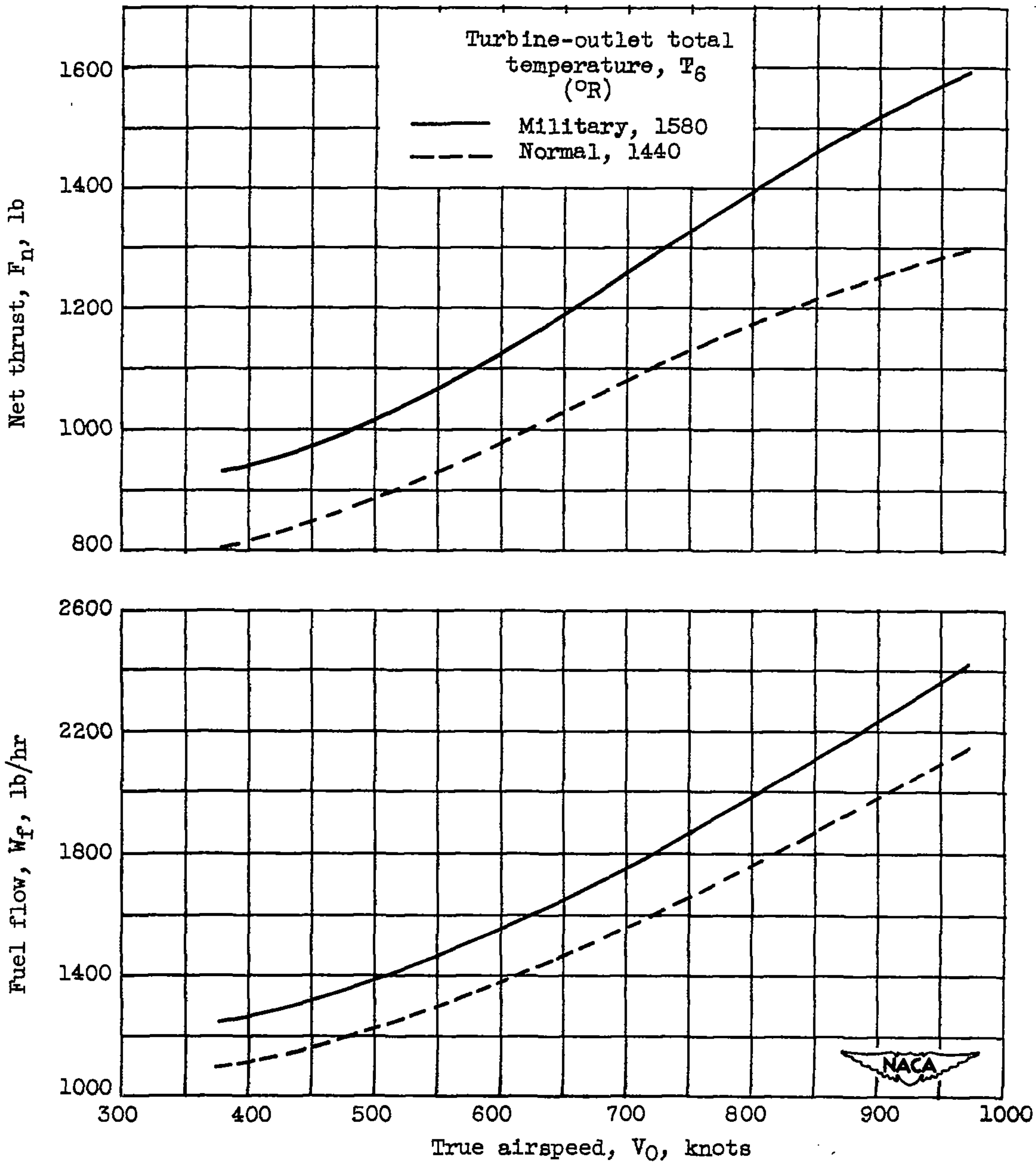

Figure 12. - Variation of net thrust and fuel flow with flight speed from experimental data. Altitude, 55,000 feet; engine speed, 7260 rpm.

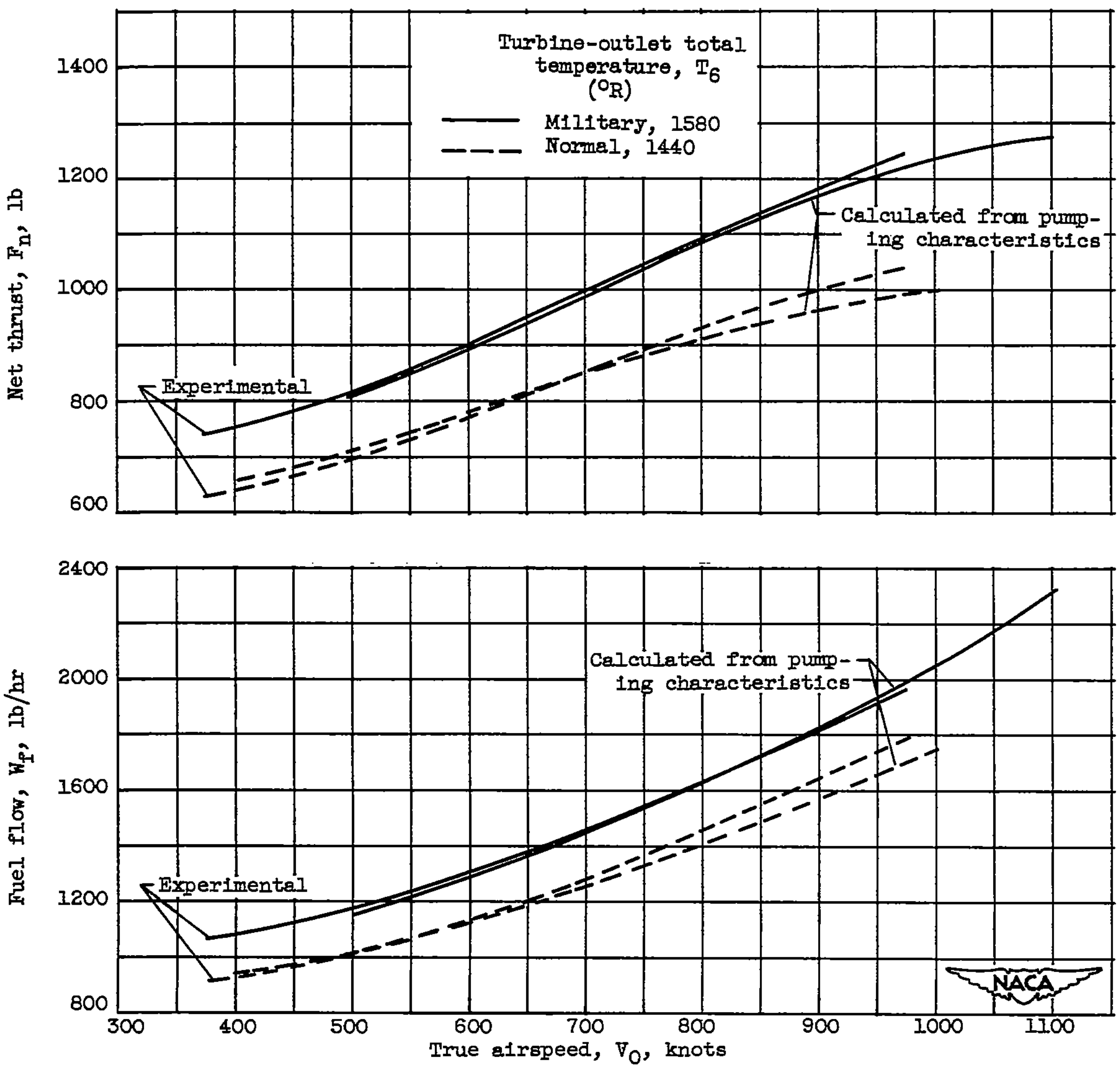

Figure 13. - Variation of net thrust and fuel flow with flight speed obtained from experimental data and data calculated from pumping characteristics. Altitude, 60,000 feet; engine speed, 7260 rpm.

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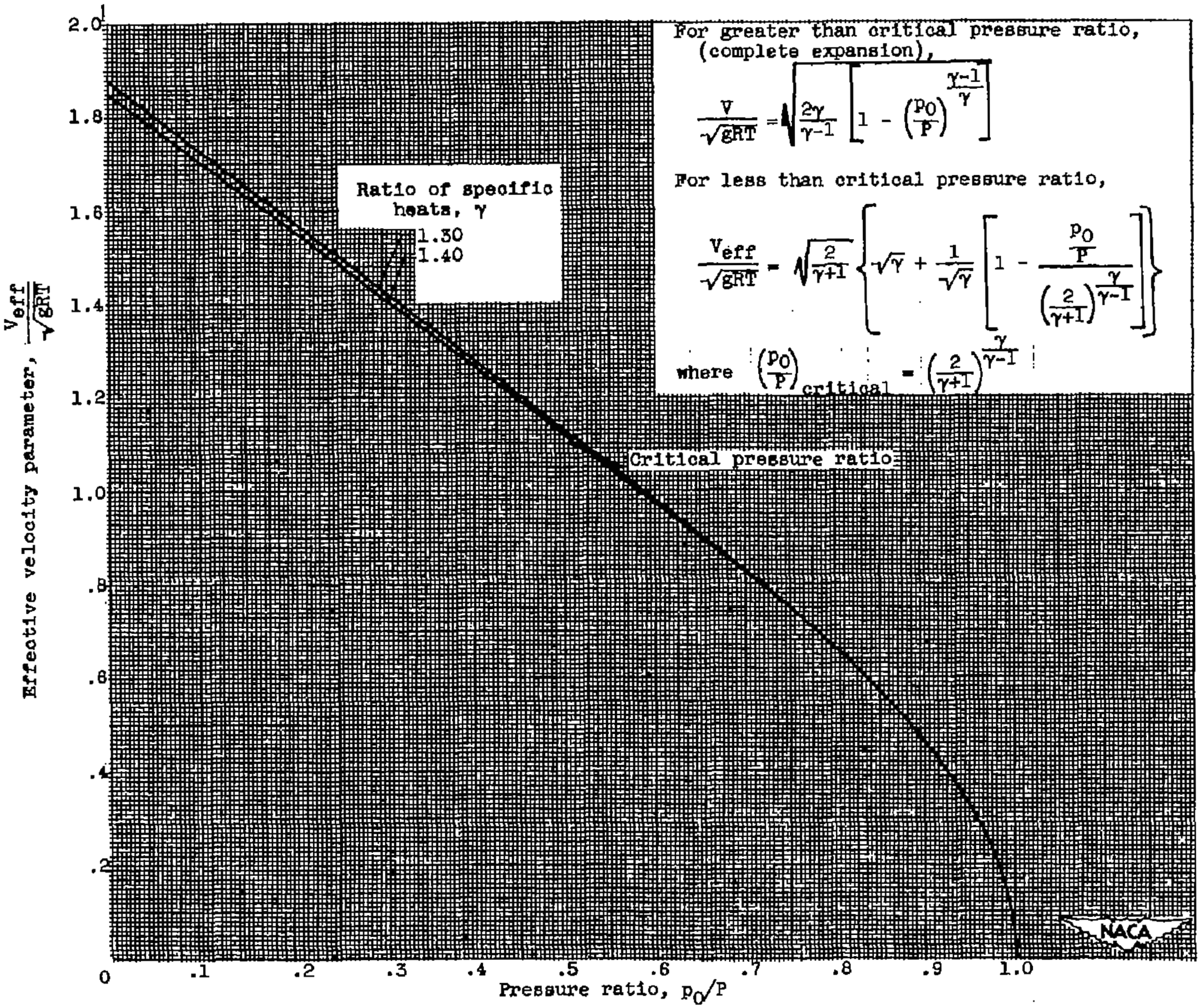

Figure 14. - Variation of effective velocity parameter with pressure ratio for convergent nozzle.

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